

University of California, Los Angeles  
Mechanical, Aerospace and Nuclear Engineering Department

**DESIGN OF A SCIENTIFIC PROBE FOR  
OBTAINING MARS SURFACE MATERIAL**

**FINAL REPORT**

*Participating Students (Term #1):*

Miles Baker  
Eric Deyerl  
Tim Gibson  
Bob Langberg  
Terrance Yee, Report Editor

*Instructor:*

R. X. Meyer, Adjunct Professor

Sponsored by NASA/USRA

June 15, 1990

(NASA-CP-14674) DESIGN OF A SCIENTIFIC  
PROBE FOR OBTAINING MARS SURFACE MATERIAL  
Final report (California Univ.) 154 p.

CSCL 037

63/91 0202109

N91-11041

Unclass

## MSP SPECIFICATIONS

OBJECTIVE: Return 1Kg martian soil sample from the surface of mars to a mothership in a 600km mars orbit.

### STRUCTURE:

overall height : 1.2 meters (touchdown configuration)

                  1.5 meters (stowed/descent configuration)

overall width : 0.66 meters (touchdown configuration)

                  1.5 meters (stowed/descent configuration)

maximum gross vehicle mass: 110.25 Kg (after mothership separation)

mass at return: 2.00Kg (includes 1Kg sample)

planform: 3-legged layout with hexagonal bus

materials: Beryllium(launch rings, sample retrieval system,  
heat shield backing, landing gear main struts)  
Magnesium (shock absorbers, landing gear secondary  
struts, bus frame)  
Aluminum (bus platform, vehicle skin)

### PROPULSION

deorbit/descent/ascent: 1 single restartable liquid propellant  
motor,pressure fed, using 50%UDMH, 50%Hydrazine  
as a fuel and nitrogen tetroxide as oxidizer

apogee kick(rendezvous): solid propellant motor

maneuvering: (8) thrusters using UDMH as propellant

## MISSION PROFILE

0 - 1

This mission begins with detachment from the mother ship, the Mars Orbiting Vehical (MOV) of the designed spacecraft, the Mars Surface Probe (MSP). After separation the MSP determines its position and the relative position of the landing sight and initiates a short burn which will cause the MSP to be at our optimum position for deorbit within seven days. During this time the ship is in a dormant mode with occasional systems checks made by a monitoring microcomputer. Adjustments to the precession orbit are periodically made. When the MSP is in the correct location to descend with a minimum fuel expendature and no plane change, the main motor fires, decelerating the ship.

1 - 2

The MSP turns around to face its heat shield toward the direction of flight. When the atmosphere is encountered at about 200-150km altitude the ship releases a kind of "drogue chute" for passive stability during descent.

2 - 3

The MSP sheds velocity during reentry, protected by its heatshield and a multilayer insulation. At 1km above the actual ground surface a parachute is deployed to slow the craft even further. The heat shield is jettisoned with explosive bolts. After slowing to 42 m/s and drifting to within 100 meters of the surface the chute is released and the main motor fires to bring the craft to a standstill 5 meters above the surface.

3 - 4

After stopping 5 meters above the surface the main motor shuts down and the MSP falls freely to avoid contamination of soil samples. Once safely on the ground a shovel type mechanism deploys and begins collecting sample soil.

4 - 5

Once the sample has been collected and stored the ship waits for half a martian day until the MOV orbit will be nearly overhead. The MSP ascent stage separates from the hexagonal descent stage leaving spent fuel tanks, the sample collector, the landing gear, and other used parts.

5 - 6

The ascent stage climbs out of the atmosphere using the main liquid motor for the last time.

6 - 7

The ascent stage burns out. The computer points the rendezvous stage in the right direction, spins it up and releases it. There is no active guidance on the rendezvous stage.

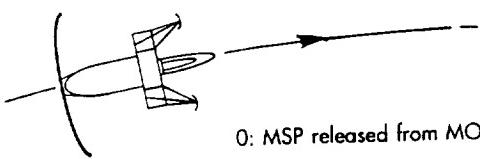
7 - 8

The rendezvous stage makes its solid motor burn and activates its beacon for the MOV to find it. Its orbit differs slightly from the MOV so the two will eventually pass each other.

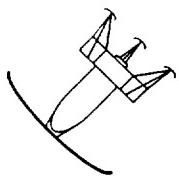
8 - 9

The MOV catches up to the rendezvous stage and brings it inside the docking bay which originally held the full MSP and then secures it for the return trip.

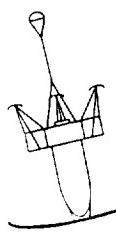
7: Solid apogee motor burn and orbital insertion



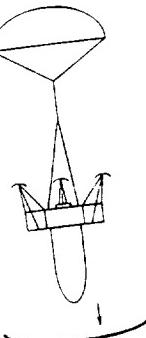
0: MSP released from MOV, main liquid motor burn



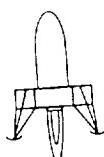
1: End of burn, reorientation of MSP



2: Descent through atmosphere, deployment of guide/drogue chute

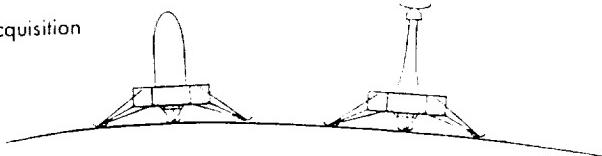


## MSP Mission Scenario



3: Chute release, main engine retro firing

5: Ascent stage separation



4: Sample acquisition

Mass:	Final (on orbit):	
	Dirt:	1.00 (kg)
	Sample Container:	0.10 (kg)
	Apogee Engine Case:	0.25 (kg)
	Beacon:	0.30 (kg)
	Batteries/solar cells:	0.15 (kg)
	Structure:	0.20 (kg)
	Total:	2.00 (kg)
	Before Separation:	
	Apogee Stage:	2.00 (kg)
	Apogee Fuel:	0.73 (kg)
	Apogee Stage Support:	0.90 (kg)
	Fuel and Helium Tanks:	3.10 (kg)
	Valves and Regulators:	0.50 (kg)
	Aeroshell:	1.80 (kg)
	Guidance and Control:	2.00 (kg)
	Batteries:	0.10 (kg)
	Liquid Engine and Plumbing:	2.00 (kg)
	Engine Structural Support:	0.90 (kg)
	Thermal Insulation:	0.50 (kg)
	Attitude Control:	0.50 (kg)
	Total:	15.03 (kg)
	Ascent Stage Before Liftoff:	
	Ascent Stage Dry Mass:	15.03 (kg)
	Ascent Fuel (5% ullage):	44.17 (kg)
	Total:	59.20 (kg)
	Mass on Mars:	
	Ascent Stage:	59.20 (kg)
	Descent Tanks and Plumbing:	1.50 (kg)
	Landing Gear:	1.50 (kg)
	Payload Scoop:	15.00 (kg)
	Batteries:	0.80 (kg)
	Laser Range Finder:	1.00 (kg)
	Landing Computer:	1.00 (kg)
	Attitude Control:	0.50 (kg)
	Thermal Insulation:	1.50 (kg)
	Structure:	7.55 (kg)
	Total:	89.55 (kg)
	Mass at Re-entry:	
	Landing Mass:	89.55 (kg)
	Retro Fuel:	3.00 (kg)
	Parachute:	2.00 (kg)
	Parachute Canister:	0.30 (kg)
	Heat Shield and Supports:	5.00 (kg)
	Drogue Basket:	2.00 (kg)
	Total:	101.85 (kg)
	Initial (On-Orbit) Mass:	
	Re-entry mass:	101.85 (kg)
	Descent Fuel:	7.57 (kg)
	Orbital Maneuvering Fuel:	0.83 (kg)
	Total:	110.25 (kg)

Table I  
MSP Mass Estimates

# **Contents**

**Section A: Structural Design and Component Placement**  
Eric Deyerl

**Section B: Thermal Control and Guidance**  
Terrance Yee

**Section C: Propulsion Systems**  
Myles Baker

**Section D: Orbital Mechanics**  
Bob Langberg

**Section E: Specialized Structures**  
Tim Gibson

Section A:  
**Structural Design and Component Placement**

Eric Deyerl

## **1. Goal**

---

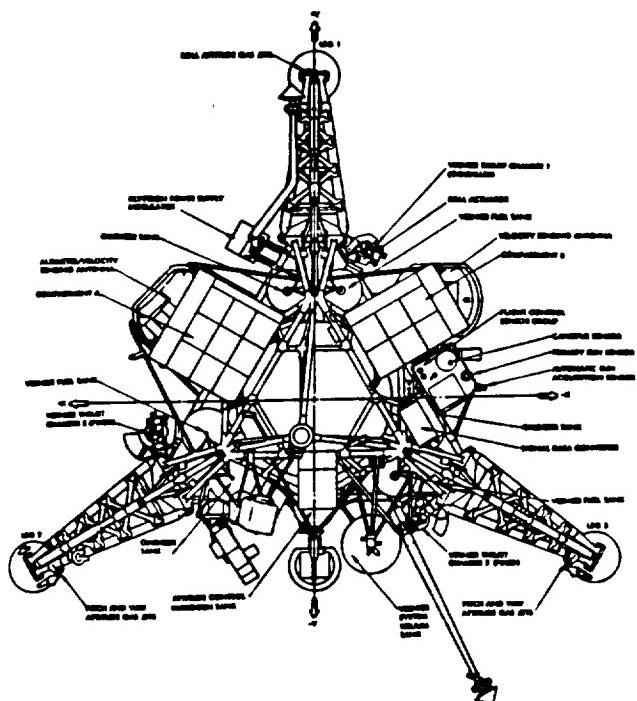
To provide the most compact (space and weight efficient) structure to unite, protect, and allow effective operation of all spacecraft components.

## **2. Development**

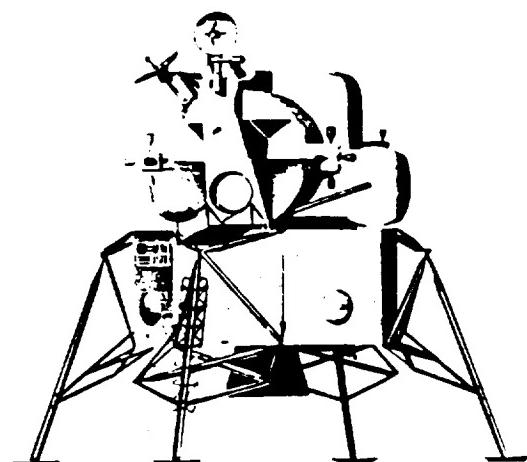
---

### **2.1 Research**

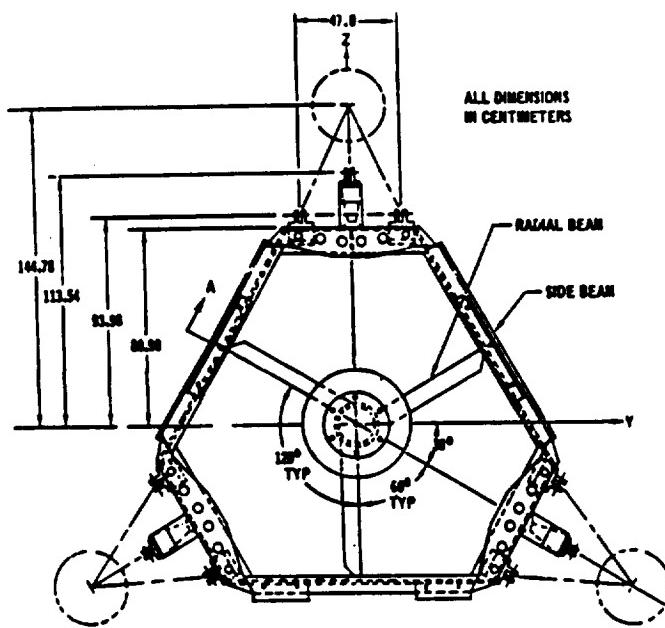
Much research was accomplished, mainly through the perusal of large amounts of NASA technical reports (see References). Here, the designer focused his attention on reports that dealt with landing gear and spacframe studies. Most specifically, there were two NASA projects that provided the most insight in these areas; those were the Surveyor and Apollo programs. (Figures 1 and 2). Although both of these missions were lunar in objective, they had similar parameters in their missions, and so were valuable in the information they provided.



**Figure 1: Surveyor spacecraft**

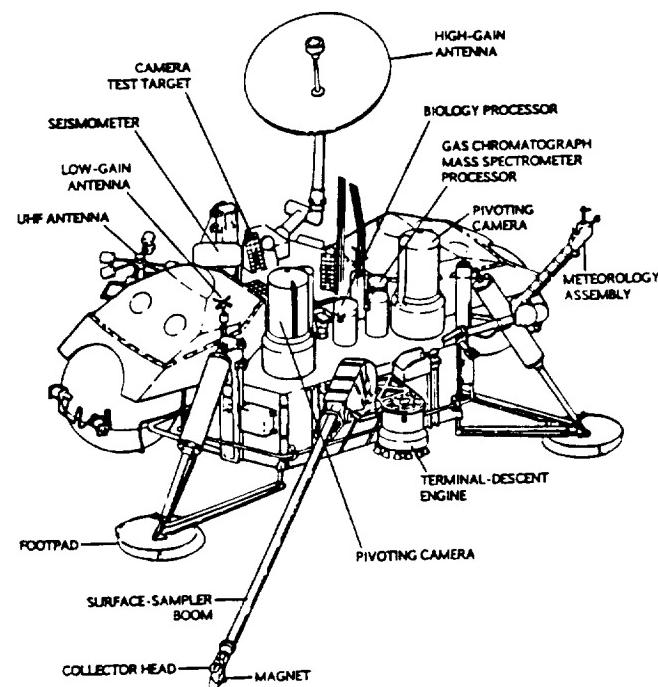


**Figure 2: Apollo lunar lander**



**Figure 3: Generic three-legged lander planform**

The surveyor three-legged layout was inspirational for the MSP designer. It is simplistic and effective, and had been used successfully in the Viking mission to Mars. (Refer to Figures 3 and 4) With three legs, one can be sure that all footpads will be in contact with the Martian soil upon landing. The three-legged design also allowed nicely for a structural-rigid hexagonal bus structure. Finally, it is also a stable platform whose shape makes it difficult for surface winds to catch a broad side of the craft and topple it.

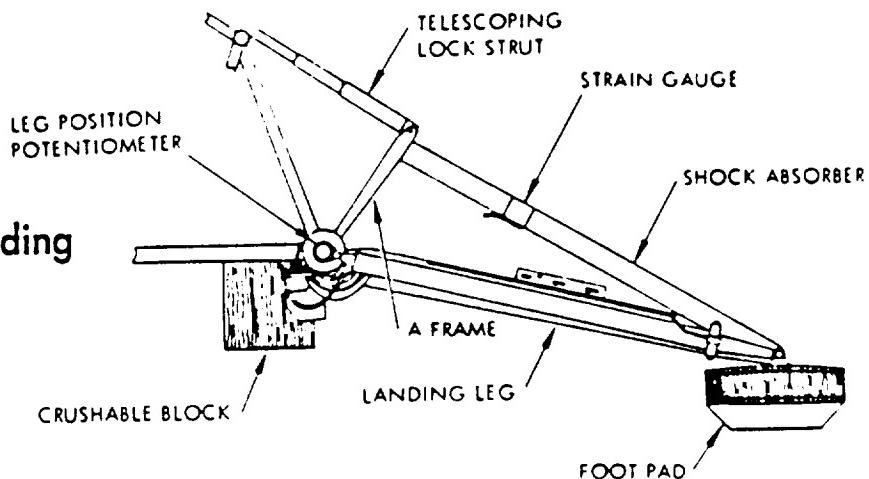


**Figure 4: Viking spacecraft**

THIS PAGE IS  
OF POOR QUALITY

The surveyor project also gave valuable insight into landing gear and shock absorber design, in conjunction with the placement of crushable honeycomb material. (Figure 5)

Figure 5: Surveyor landing gear leg



**LANDING GEAR STUDY:**  
• Contaminated Shock Absorber  
(LM-type)

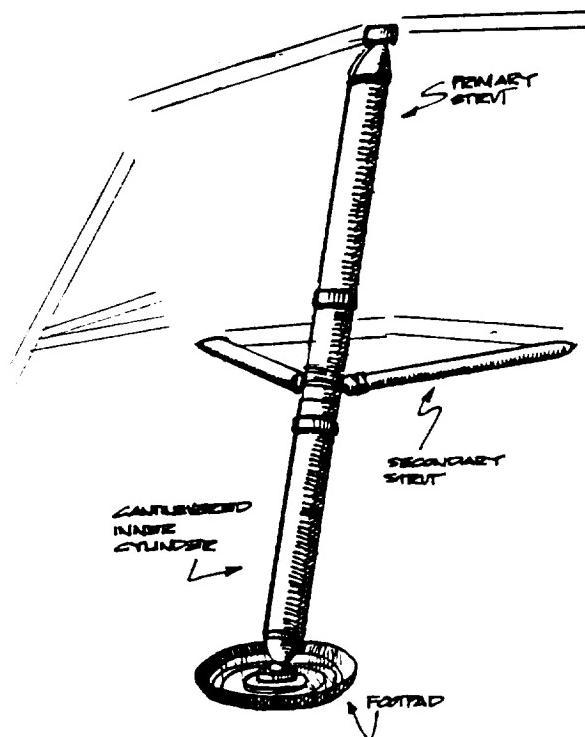


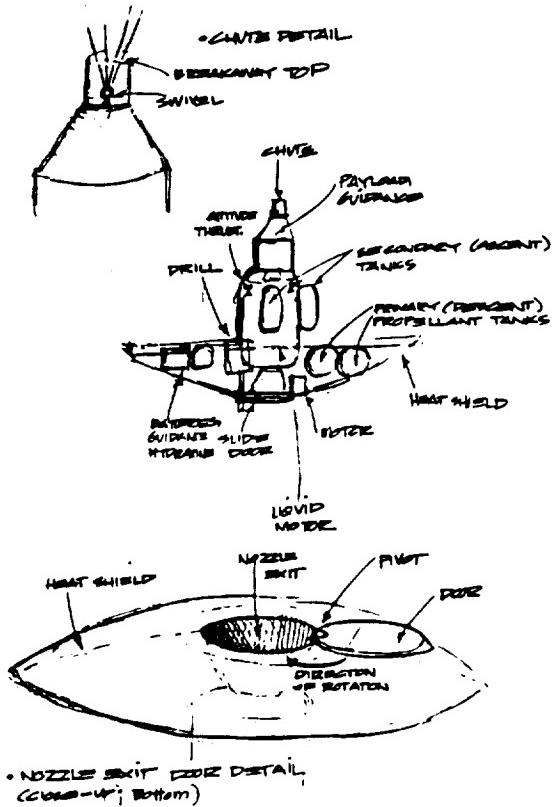
Figure 6: Apollo lunar lander landing gear leg

The Apollo Lunar Module's landing gear was also studied in an effort to incorporate a similar retracting system on the MSP. This was later found unnecessary. A rendition of one of the "Eagle's" gear is found in Figure 6.

## 2.2 Preliminary Layout Studies

Prior to much of the research, preliminary sketches were made to seek out viable spacecraft layouts. Two early plans and some notes on each are given in Figures 7 and 8.

### □ PRELIMINARY MSP LAYOUT I



### □ PRELIMINARY MSP I: COMMENTS

#### □ ASSETS:

1. Use of liquid main engine eliminates need for separate de-orbit, descent, and ascent solid motors
2. Broad-based heat shield allows for surface discontinuities, thus eliminating need for more complex landing structures (legs, shock absorbers, etc.)
3. Throttatable main engine can aid in descent so parachute may be cut early, preventing any interference with MSP.
4. Large shield allows for mounting of much hardware that can be left behind on Mars

### □ LIABILITIES:

1. Use of single liquid engine may incur weight penalties in form of additional hardware (tanks, plumbing, pumps)
2. Door in heat shield may not be feasible due to complexity and possible imperfect sealing.
3. Use of broad shield for landing may make difficult to land on a surface with large irregularities.

Figure 7: First preliminary MSP layout sketch and comments

Figure 8: Second preliminary MSP layout sketch and comments

### □ PRELIMINARY MSP II: COMMENTS

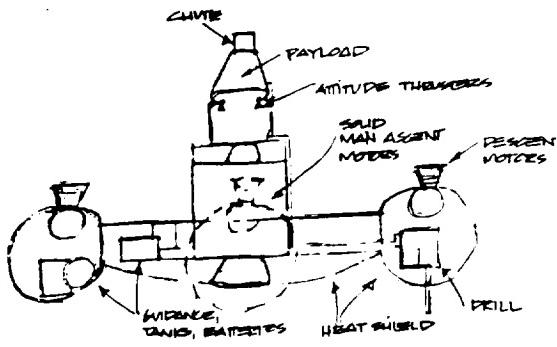
#### □ ASSETS:

1. Use of several small 'descent' motors eliminates need for one large main.
2. Tri-pod type contact points assure contact with surface at all points
3. Use of solid ascent motors eliminates need for liquid plumbing, tanks, pipes

#### □ LIABILITIES:

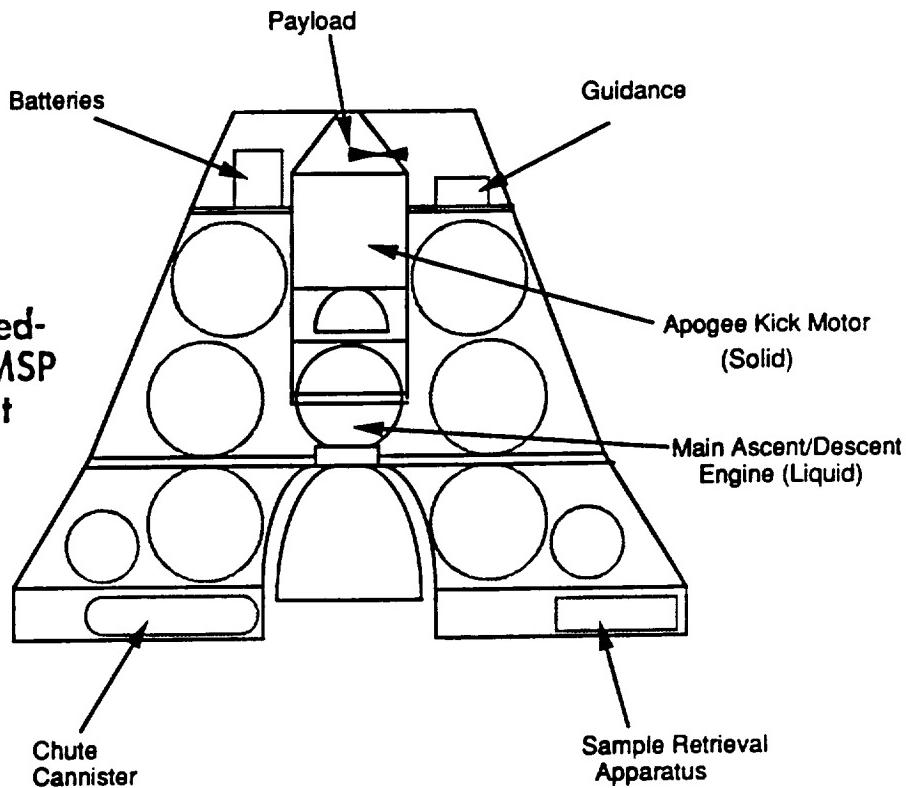
1. Possible interaction of solid descent motor outgassing with probe components.
2. Layout somewhat large; not best use of quantity of material

### □ PRELIMINARY MSP II LAYOUT



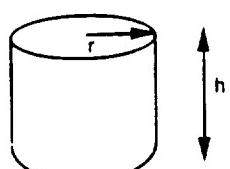
Later, after more propulsion studies and relative sizing of components had been undertaken, it was possible to produce more accurate and intermediate layouts, as shown in Figure 9.

**Figure 9: Intermediate MSP layout**



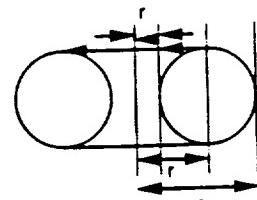
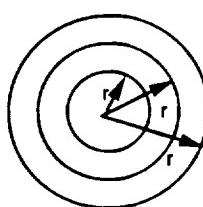
It was at this point that the designer realized that the critical components, as far as space was concerned, were the fuel and oxidizer tanks. (which is usually the case with light aerospace vehicles). Thus, a fuel tank study was initiated, which (as shown in Figure 10) resulted in a possible toroidal tank configuration. This idea was later discarded due to complexity of purging, pressure containment, and the like.

**CYLINDERS**



Volume = $r h = 17,200 \text{ cc}$	
$r$	$h$
10	55
15	24
20	25
25	9

**TOROIDS**



$$\text{Volume} = [ ((r - R)/ ) ] [ ((R + r)/ ) ]$$

(Approximation based on  $r = R$ )

**Figure 10: Fuel tank study**

$r$	$R$
8	26
10	25.5
12	25.25
15	24

## 2.3 Final MSP Configuration

After many hours of conference, deliberation, and improvisation, the layout of the MSP was finalized to that shown in Figures 11 and 12.

It is a space efficient design, and yet tries to remain as simple as possible. The design is roughly axisymmetric, with the ascent vehicle able to clear all bus components.

## 3 Component Design

Following are the steps taken to design the critical components of the MSP, which include the landing gear, shock absorbers, bus frame and platform, and ascent and apogee stage launch rings.

Care was taken to use the lightest and (naturally) strongest materials possible, with thought given to their manufacture and interaction with the environments the MSP would be subjected to. It also was always desired to use a safety factor of at least 1.5. As will be shown, because of the generally small dimensions of the craft, this would never be a problem.

### 3.1 Landing Gear

It was chosen to use the landing gear configuration shown in Figures 13 and 14, with a main strut containing a shock absorber to cushion the vehicle at landing, and secondary struts for any torsional forces encountered by the gear. The following are the calculations made in the design of these components.

#### 3.1.1 Loads on Gear

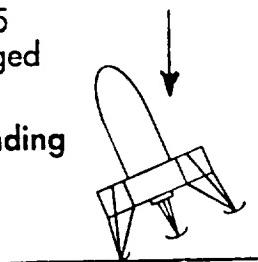
The mission scenario calls for the MSP to impact the Martian surface at 5.5 m/s. The force on one gear (considered in the extreme case of a one-legged landing) is given by

Figure 15: One-legged landing

$$F = ma$$

where  $m = 73.9 \text{ kg}$  (at landing)  
 $a = (v^2/2s)$

where  $v = 5.5 \text{ m/s}$   
 $s = 0.20 \text{ m}$  (vertical gear travel)



Thus, the force on one leg (taken to be applied through the axis of the shock absorber in the case of a lopsided one-legged worst-case landing, depicted in Figure 15) is

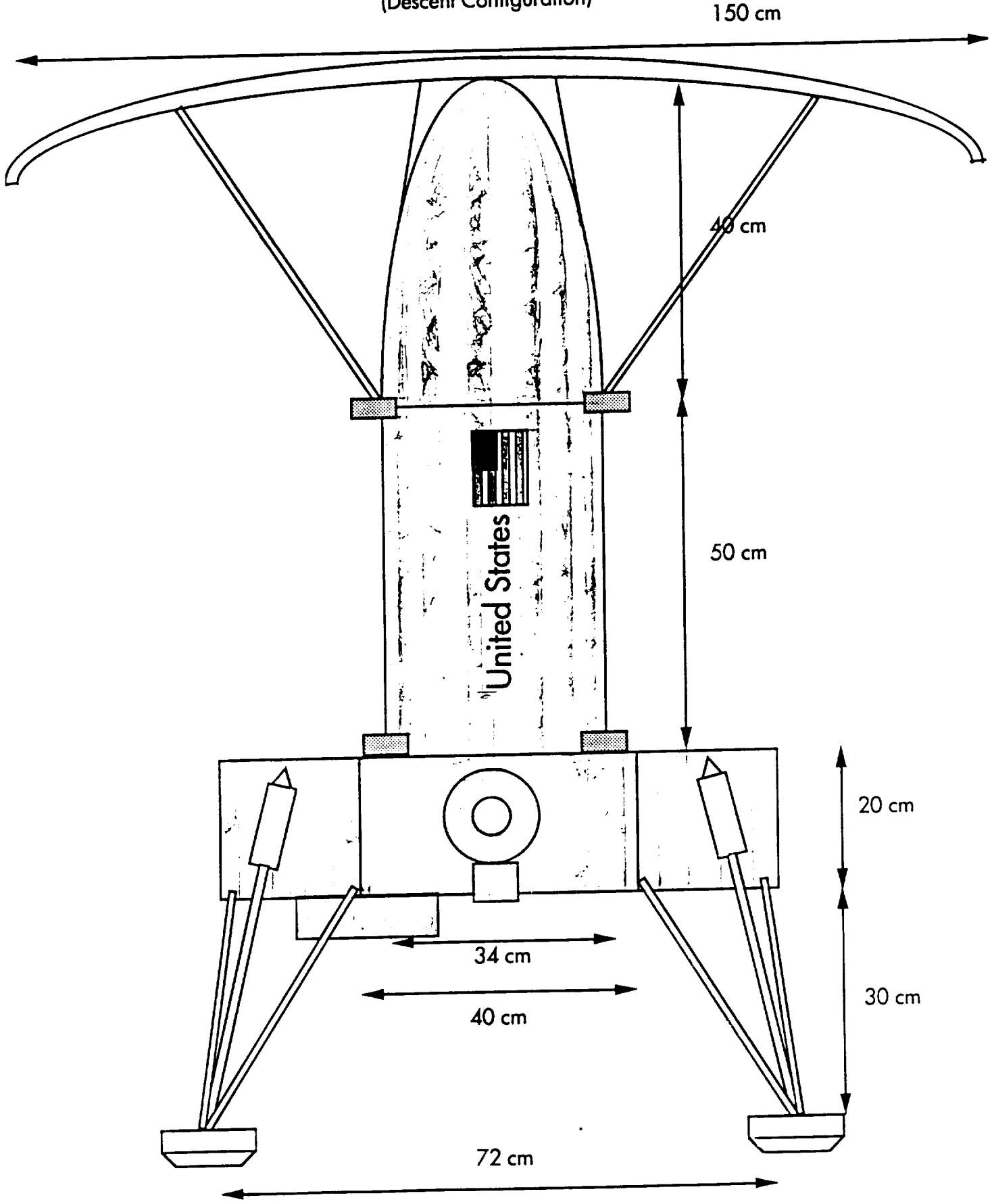
---

Landing force on one leg =  $F = 5589 \text{ N}$

---

**Figure 11.1:MSP Exterior Dimensions**

(Descent Configuration)



# Figure 11: MSP Cutaway

(Descent Position)

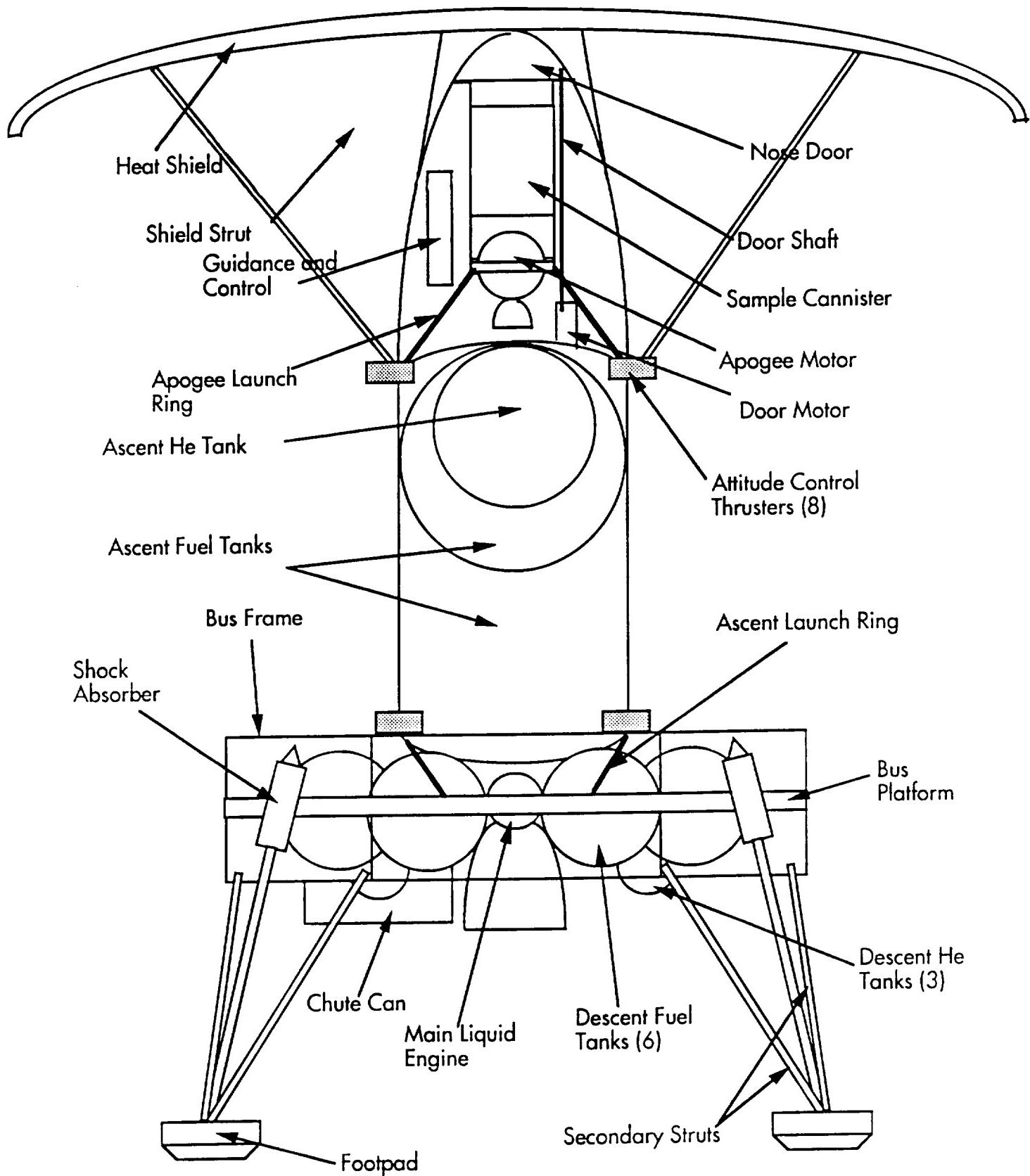


Figure 12.2: MSP Planform Exterior

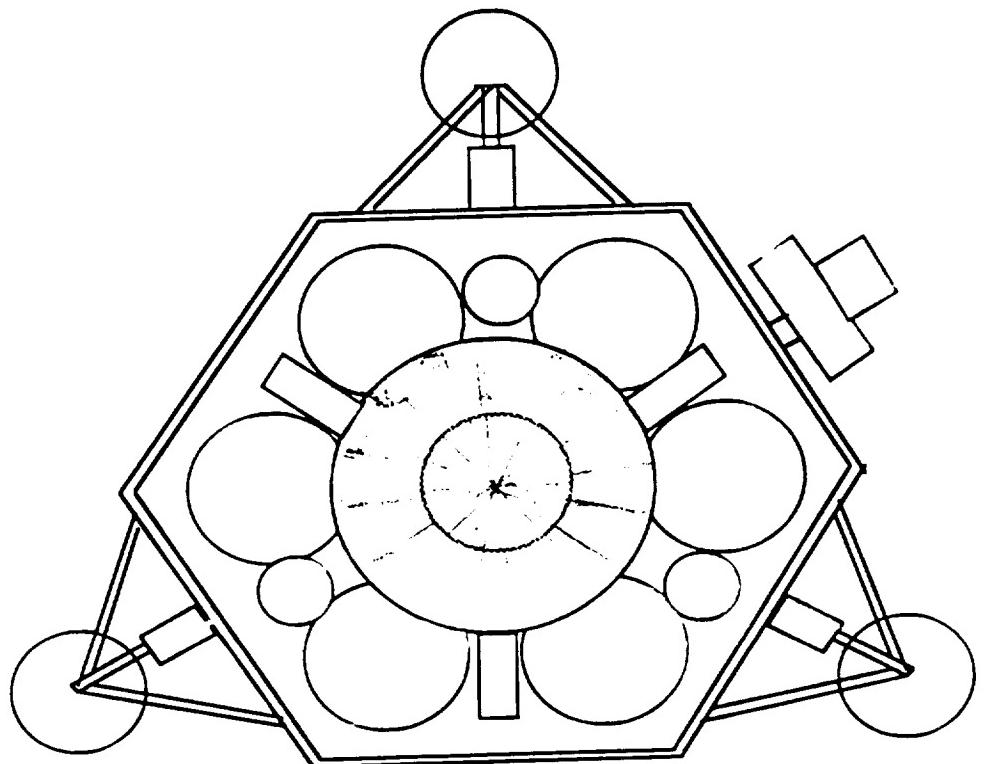
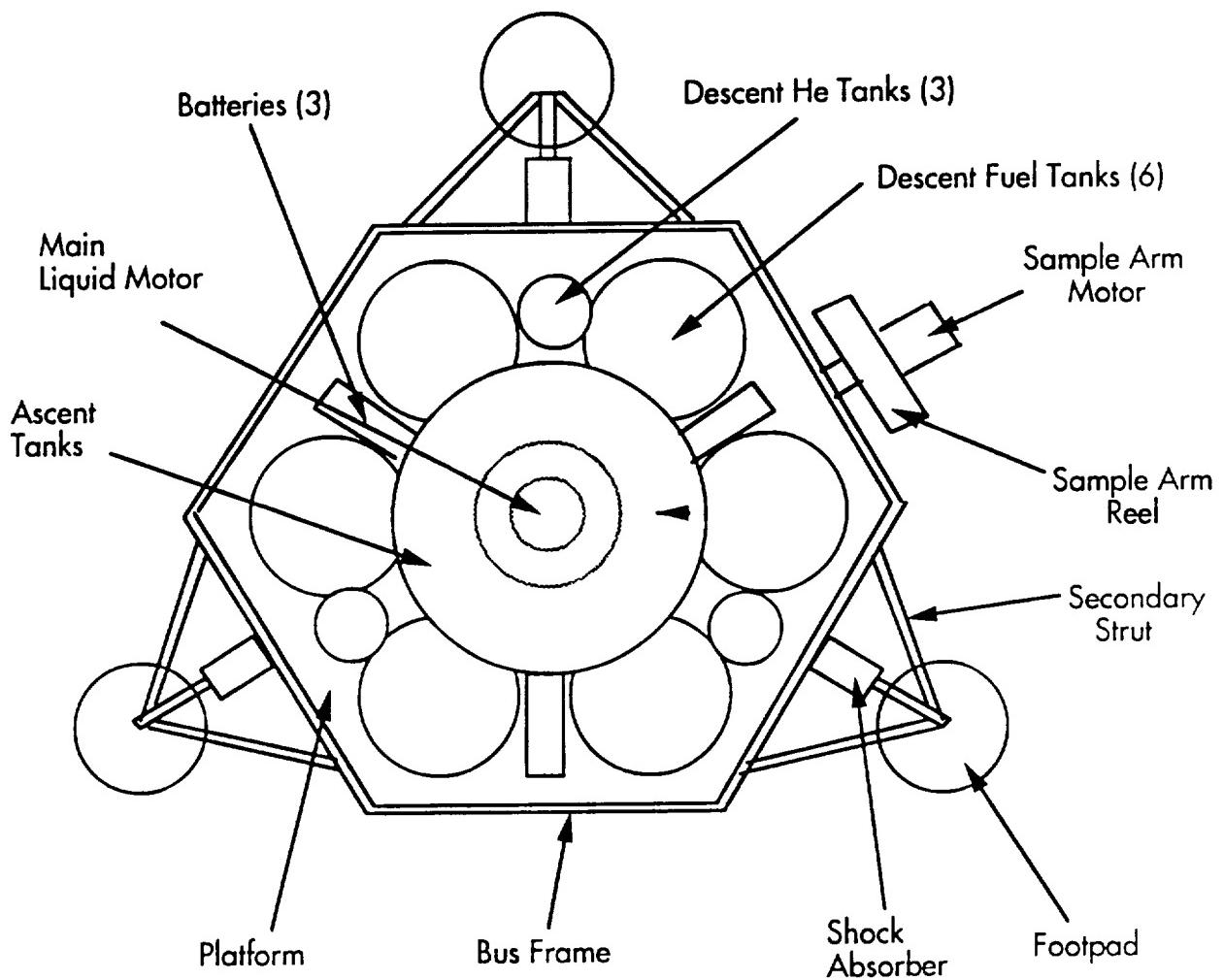
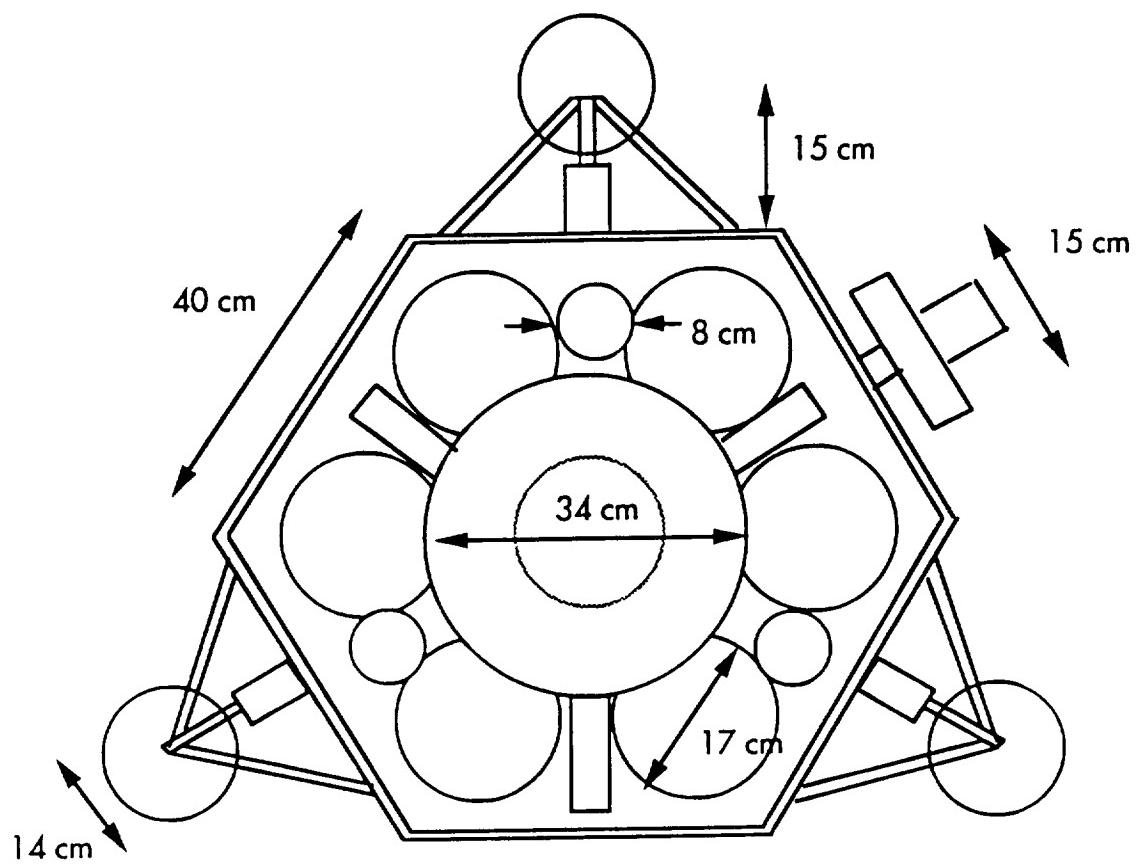


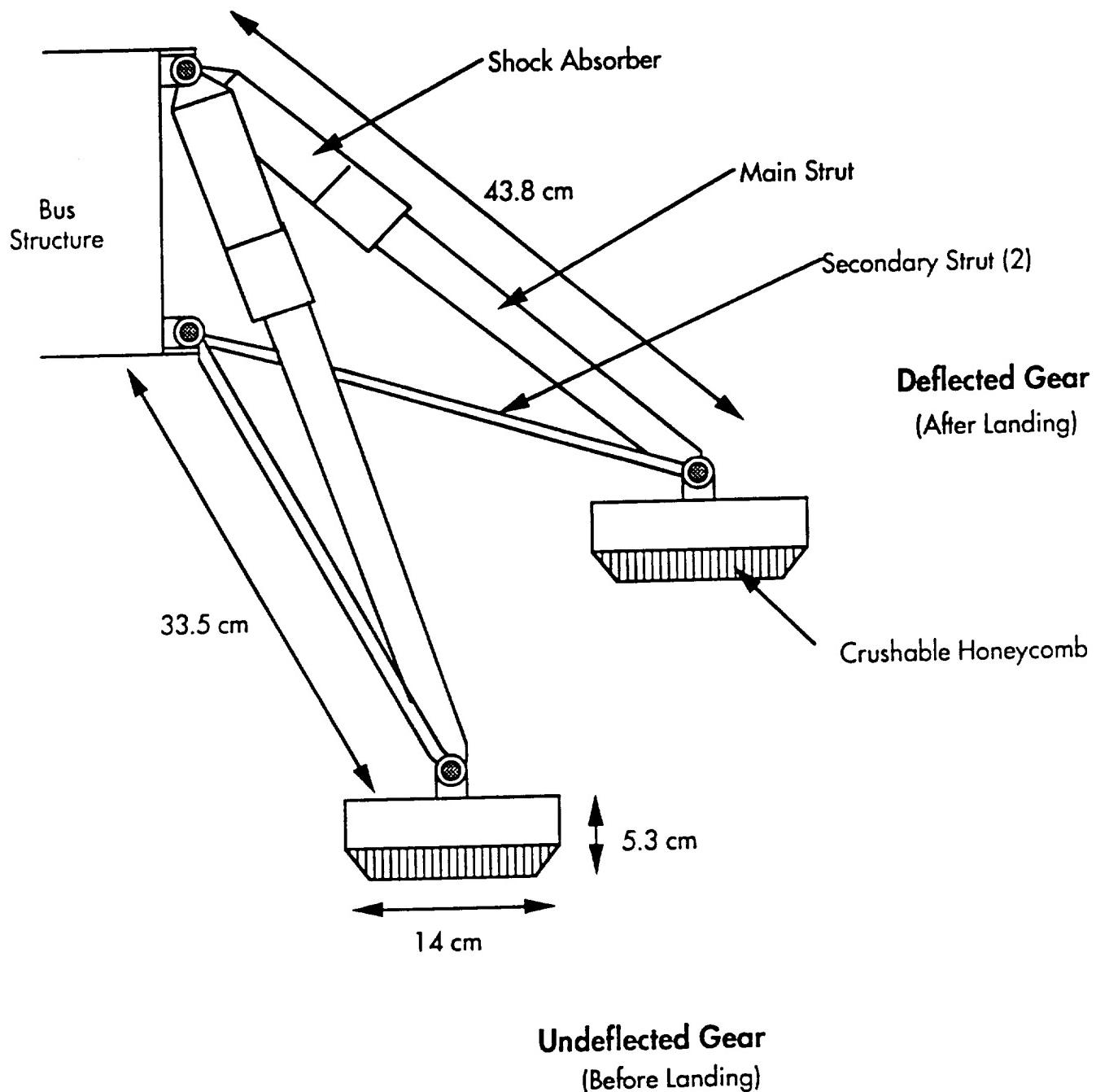
Figure 12: MSP Planform Cutaway



**Figure 12.1: MSP Planform Dimensions**



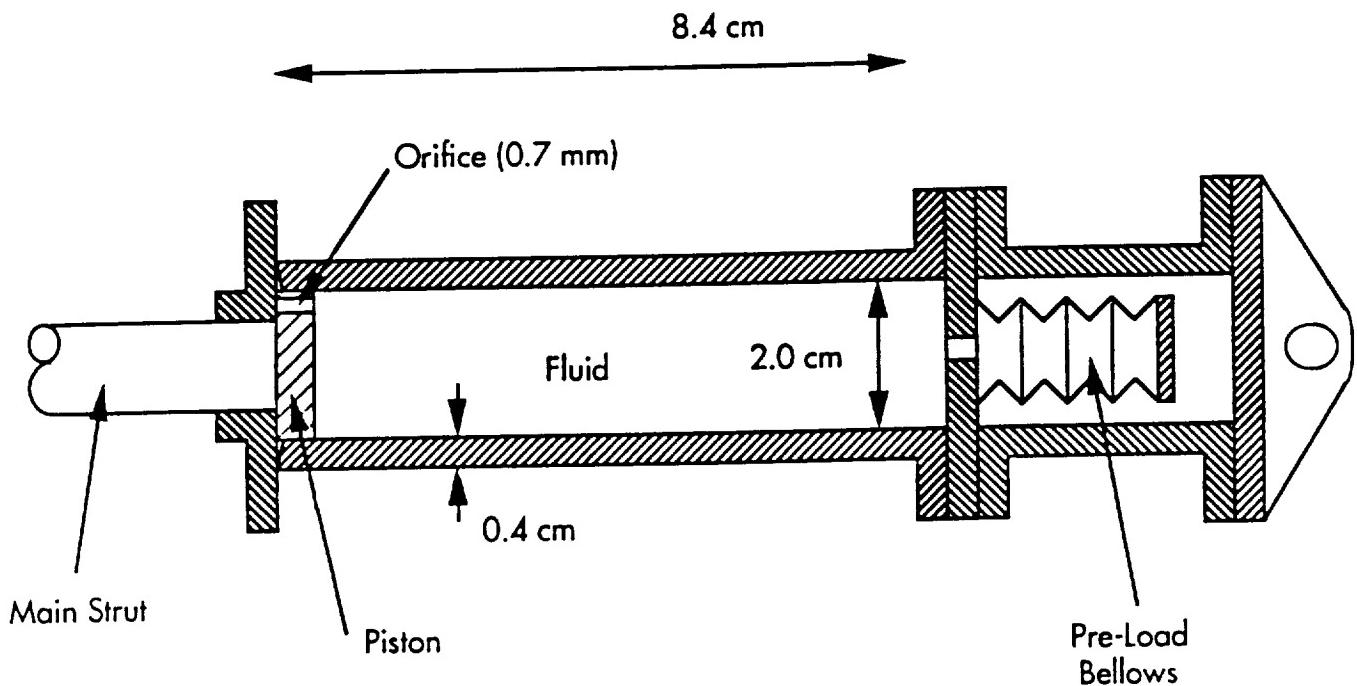
**Figure 13: Landing Gear Geometry**



### 3.1.2 Shock Absorber

The shock absorber chosen was a fluid-filled type, in which the travel of the piston forces fluid through a small orifice in the piston, damping the spacecraft's motion. A cutaway of the MSP shock is given in Figure 16.

**Figure 16: Shock Absorber Cutaway**



From the derivation given in class, which uses Bernoulli's principle to equate mass flow generated by the motion of the piston of area A to the resulting mass flow through the orifice of area  $A^*$ , the following relation results:

$$A/A^* = (2F/\rho A)^{1/2} (1/v)$$

where  $F = 5589 \text{ N}$  (landing force)  
 $\rho = 917 \text{ kg/m}^3$  (assumed from SAE 30W oil)  
 $v = 5.5 \text{ m/s}$  (landing velocity)

which results in

$$A^* = (1.57)A^{3/2}$$

Then, for the sake of a compact design, choosing a piston radius of 1 cm gives

---

Piston area =  $A = 3.14 \times 10^{-4} \text{ m}^2$   
Orifice area =  $A^* = 8.74 \times 10^{-6} \text{ m}^2$

---

From the geometry of the landing gear, it can be seen that to effect a 20 cm vertical deflection of the footpad, it is necessary to have

---

Piston stroke travel = 8.4 cm

---

To design the dimensions of the shock cylinder for strength, it was necessary to find the maximum pressure developed in the cylinder. Here, it was assumed that immediately upon impact, the maximum force of landing was transmitted to the shock absorber (this is obviously a more-stringent-than-reality assumption). So here, the pressure is given by

$$P = F/A$$

where again       $F = 5589 \text{ N}$   
                       $A = 3.14 \times 10^{-4} \text{ m}^2$

---

Maximum pressure in shock cylinder = 43.76 atm  
( $44.36 \times 10^5 \text{ Pa}$ )

---

Finally, to find the maximum stresses occurring in the cylinder, we use the relations from Ugural,

$$\sigma_r = (r_i^2)(P)(1 - (r_o^2/r_i^2))/(r_o^2 - r_i^2)$$

$$\sigma_\theta = (r_i^2)(P)(1 + (r_o^2/r_i^2))/(r_o^2 - r_i^2)$$

$$\sigma_z = F/(p)(r_o^2 - r_i^2)$$

where here,       $r_i = 0.01 \text{ m}$   
                       $r_o = 0.014 \text{ m}$   
                       $P = 44.36 \times 10^5 \text{ Pa}$

This results in the following

---

Maximum stresses in the shock cylinder

---

$$\sigma_r = 4.44 \text{ MPa}$$
$$\sigma_\theta = 24.59 \text{ MPa}$$
$$\sigma_z = 18.53 \text{ MPa}$$

---

As for the material chosen, at first Lockalloy, the Be-Al metal looked promising, but proved too heavy for original gear weight estimations, so the following metal was chosen (Bruhn):

---

**Material for Landing Gear: AZ61A Magnesium**  
**Ultimate Strength: 96.53 MPa**  
**Density: 1791 kg/m<sup>3</sup>**

---

Obviously, our safety factors are well-satisfied, and as long as the Magnesium can be machined to 4 mm, there should be no problem. In fact, the weight savings are tremendous:

**Mass of shock absorber:**

$$\text{Fluid} = (\text{stroke})(\text{area})(\text{density}) = (0.084)(0.000126)(917) = 97.1 \text{ g}$$
$$\begin{aligned}\text{Cylinder} &= (\text{length})(\text{area})(\text{density}) = (0.10)(\pi)(0.012^2 - 0.01^2)(2100) \\ &= 29.0 \text{ g}\end{aligned}$$
$$\text{TOTAL} = 0.189 \text{ kg}$$

In addition, the main strut must be checked for Euler buckling, where the critical load applied to the strut is given by

$$P_{cr} = \pi^2 EI/L^2$$

After performing this analysis once, it was found necessary to use Beryllium, where

---

**Material for Landing Gear Main Struts: Lockalloy (Be-Al)**  
**Young's Modulus:  $2.9 \times 10^{11}$  Pa**  
**Yield Strength:  $3.24 \times 10^8$  Pa**  
**Density: 1826 kg/m<sup>3</sup>**

---

$$E = 2.9 \times 10^{11} \text{ Pa}$$
$$I = \pi r^4/4 = (\pi)(0.01)^4/4 = 7.85 \times 10^{-9} \text{ m}^4$$
$$L = 0.40 \text{ m}$$

And thus

---

**Critical buckling load for main strut = 140,400 N**

---

Which is well within the factor of 3 for safety used for most buckling and instability cases, considering that the applied load will be about 5900 N.

**Mass of gear struts:**

$$\begin{aligned}\text{Braces (2)} &= (2)(\text{length})(\text{area})(\text{density}) = (2)(0.335)(\pi)(0.005^2)(1791) \\ &= 94.2 \text{ g}\end{aligned}$$

$$\text{Main Strut} = (0.422)(\pi)(0.012)(1826) = 229 \text{ g}$$

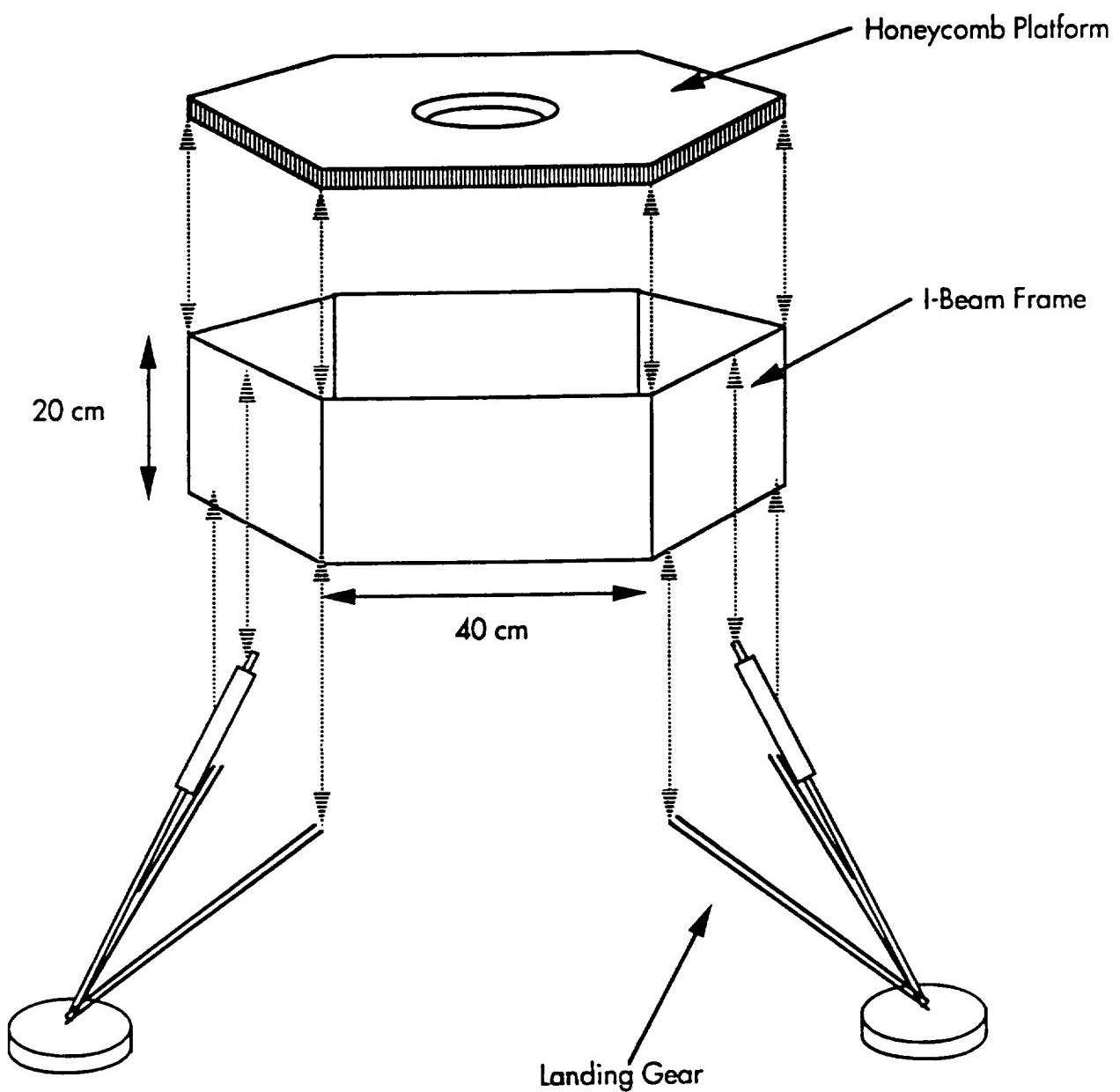
$$\text{TOTAL/leg} = 0.323 \text{ kg}$$

---

**Total mass of landing gear =  $(0.0189 + 0.154 + 0.0942)(3) + \text{footpads} = 1.50 \text{ kg}$**

---

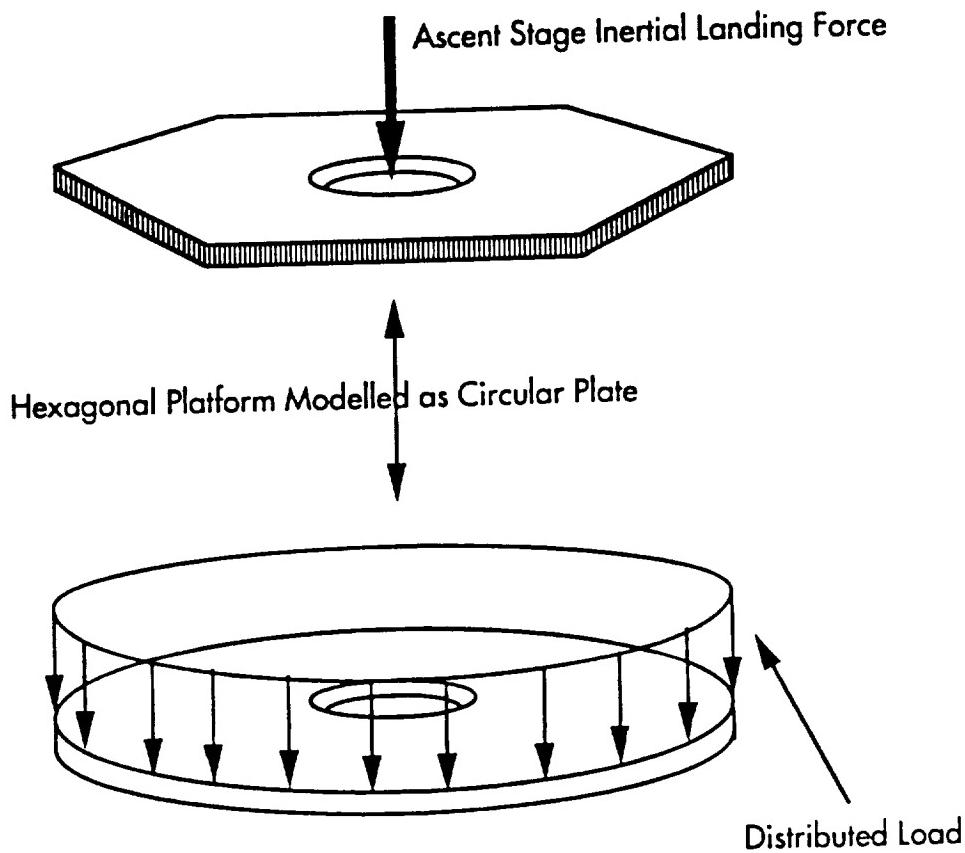
**Figure 17: MSP Bus Construction**



### **3.2 Bus Analysis**

The construction of the main structure of the MSP is, as given in Figure 17, a hexagonal frame of I-beams with a center platform made from a composite hexagonal sandwich platform. The following is the analysis done for its design.

**Figure 18: Analytical Modelling of Bus Shelf**



#### **3.2.1 Platform Design**

The bus platform encounters its main loading condition upon landing, when the inertial force of the ascent vehicle stack impinges upon the thrust ring at its center, as shown in Figure 18. Because of its hexagonal shape, the platform was modelled as a circular plate under a distributed loading, as shown in Figure 18. From Urugal, we have

$$\sigma_{r\max} = (3/4)(P_o)(r^2)/(t^2)$$

where  $P_o = \text{load/area} = (m_{disc})(a)/(\pi)(r^2)$   
 $= (45\text{kg})(75.6\text{m/s}^2)/(\pi)(0.32\text{m}^2)$   
 $= 33.2 \text{ kPa}$   
 $t = \text{thickness of plate} = 0.025 \text{ m}$

Thus  $\sigma_{max} = (0.75)(33,200)/(1 \pi) = 7.64 \text{ kPa}$

From Bruhn, we are able to find a honeycomb sandwich material as shown in Figure 19:

---

Material for platform: Al 5056 honeycomb  
Flexural rigidity: approx 4.5 MPa  
Density: 76.4 kg/m<sup>3</sup>

---

So, for our needs, the platform will have a mass of

---

Platform mass = 0.71 kg

---

And obviously, well within the range of safety.

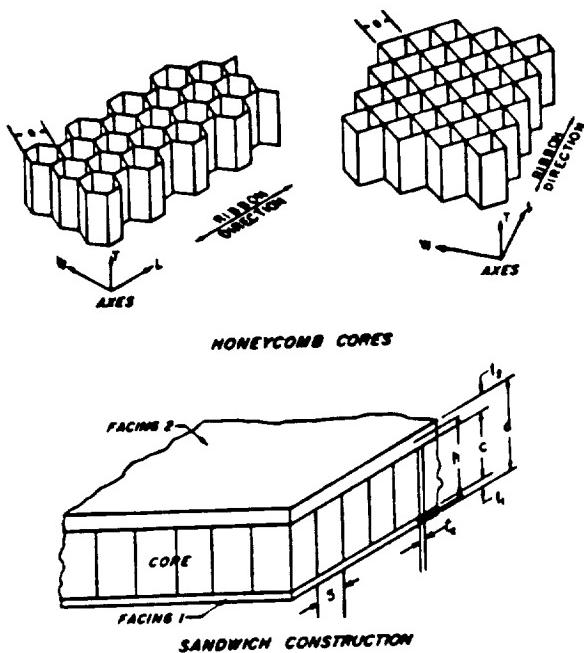
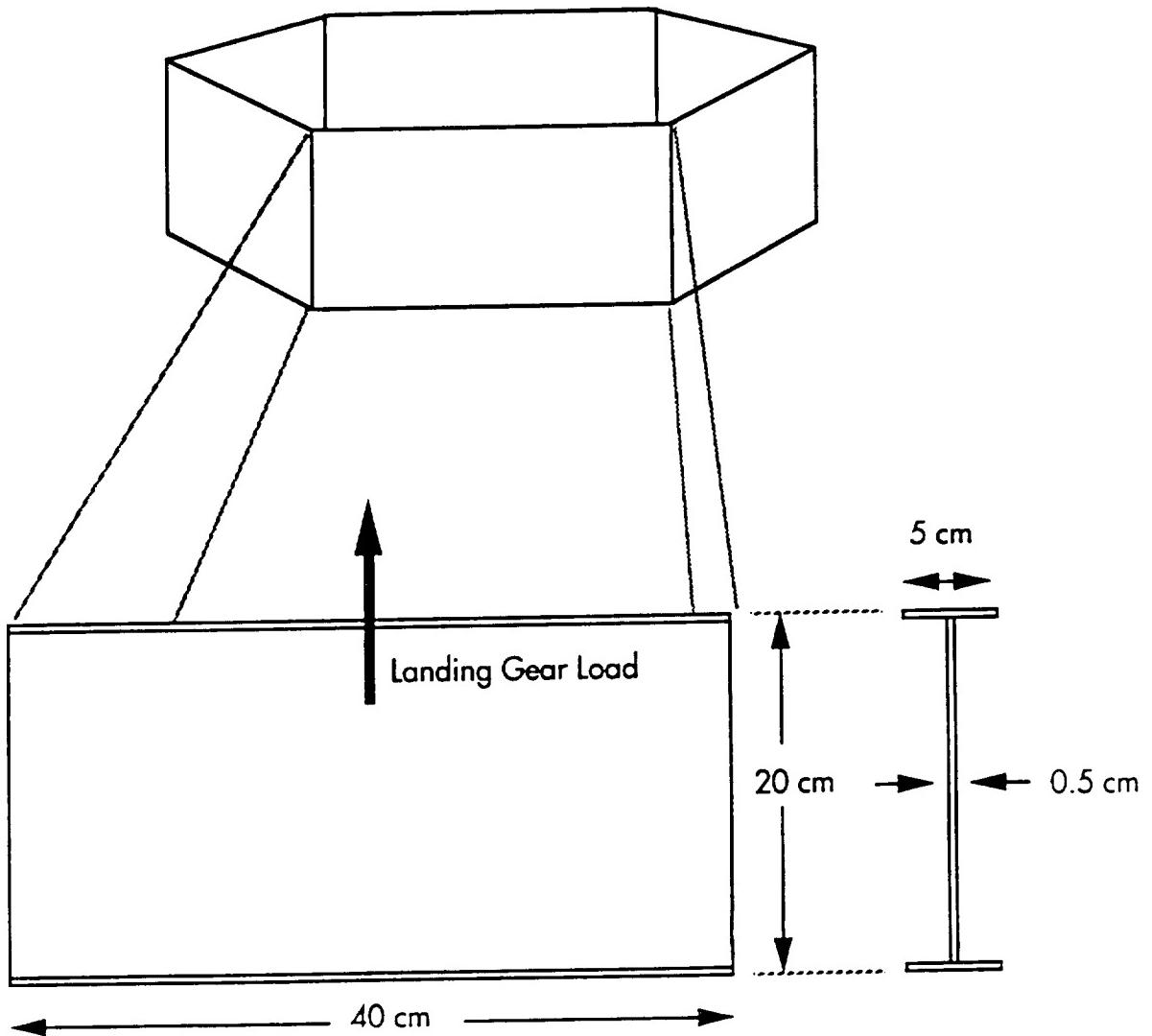


Figure 19: Platform material construction

### 3.2.2 I-Beam Design

The frame of the bus is constructed of I-beams, depicted in Figure 20. Here, the designer assumed a worst-case condition, in which a side beam is load with the total vertical force of landing transmitted through the shock absorber into the I-beam at midpoint.

**Figure 20 : Analytical Modelling of Bus Frame I-Beams**



From the classical beam theory analysis, the maximum stress occurs from bending at midpoint in the beam, and is given by

$$\sigma_b = M(h/2)/I$$

Where  $M$  = moment applied = (landing force)(half of beam length)  
 $= (5589\text{N})(0.20\text{m}) = 1118 \text{ Nm}$   
 $h$  = height of beam = 0.20 m

And where  $I$  for the beam is calculated by taking the moments of area each of the portions in Figure 21 separately, as is classically done. So, using the notations in Figure 21, we have

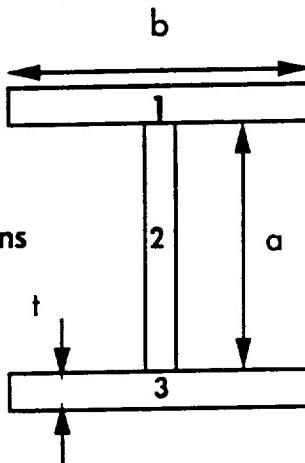


Figure 21: I-Beam Cross-Section Dimensions

Portion	Area	Arm	$(\text{Area})(\text{Arm}^2)$	<u>Own i</u>
1	$(t)(b)$	$(0.5)(a+t)$	$(t)(b)(1/4)(a+t)^2$	$(b)(t^3)/12$
2	$(t)(a)$	0	0	$(t)(a^3)/12$
3	same	as portion 1		

Inserting all the relevant dimensions gives

$$I_{\text{section}} = 7.61 \times 10^{-6} \text{ m}^4$$

Which ultimately leads to

---


$$\text{Maximum stress in beams} = 14.7 \text{ MPa}$$


---

Which is well below the same ultimate stress level of the Magnesium. As for the masses of the beams, we can use that

$$\begin{aligned} \text{Mass} &= (\text{cross-sectional area})(\text{length})(\text{density}) \\ &= [(2)(t)(b) + (a)(t)](L)(\rho) \\ &= [(2)(0.005)(0.05) + (0.19)(0.005)](0.40)(1791) \\ &= 1.04 \text{ kg/ beam} \end{aligned}$$

---

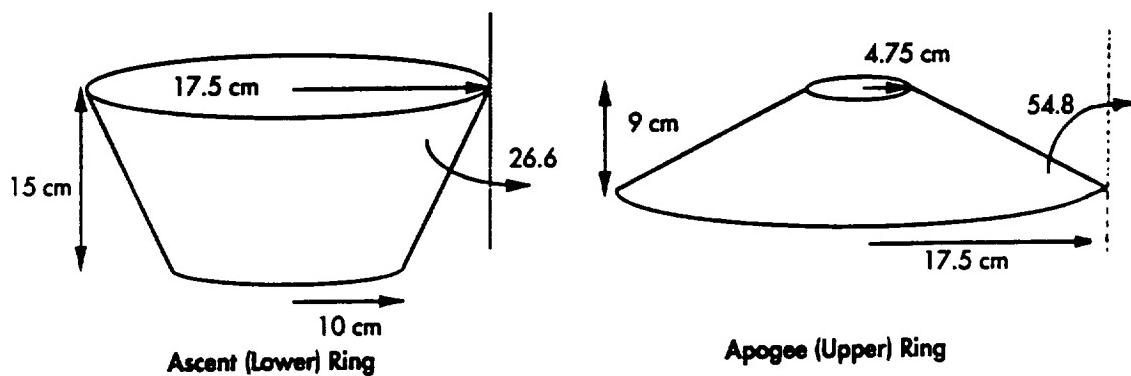

$$\text{Total mass of frame} = 6.24 \text{ kg}$$


---

### 3.3 Launch Rings

The launch rings are located as shown in the cutaway of Figure 11. The lower, or ascent launch ring is designed to join the ascent stage to the bus frame, and the upper, or apogee launch ring is designed to join the apogee stage to the ascent stage. Both rings are to be strong enough to keep the craft together upon the two major vehicle forces of chute deployment and landing. The two rings are depicted in Figure 22.

**Figure 22: Launch Rings**



Naturally, the chute deployment is a condition that puts the rings in tension, while the landing phase puts both in compression. Roughly, both need to withstand joining the vehicle at 10g's in tension and 7.5g's in compression. Because the rings separate parts of the vehicle that have different masses, the forces each experiences will be different under the different conditions.

The resulting applied stress on each ring will be given by

$$\sigma_{app} = F/A$$

where  $F = ma$

where  $m$  = mass of the portion of the space-craft supported by the ring in question under load  
 $a$  = acceleration in question

The designer then compared this resulting applied stress to the critical stress for the case of a loaded truncated cone, which is given by

$$\sigma_{cr} = (\gamma)(E)(t)(\cos\alpha)/(r)[3(1 - v^2)]^{1/2}$$

where

$\gamma$  = buckling safety factor = 0.33  
 $E$  = Young's modulus =  $2.9 \times 10^{11}$  (Beryllium)  
 $t$  = thickness of cone = 0.005 m  
 $v$  = Poisson's ratio = 0.33 (non ferrous)  
 $r$  = smaller radius  
 $\alpha$  = cone half angle

The results for different loading conditions for the two Beryllium rings of Figure 22 are summarized as follows:

Ring	$r$ (m)	$\alpha$ (deg)	Loading (g)	Supported Mass (kg)	$\sigma_{app}$ (Mpa)	$\sigma_{cr}$ (Mpa)
Upper	0.0475	54.8	7.5 comp. (landing)	2.75	0.28	3550
Lower	0.100	26.6	7.5 comp. (landing)	45.0	2.17	2620
Upper	0.0475	54.8	10.0 tens. (chute)	87.0	11.44	3550
Lower	0.100	26.6	10.0 tens. (chute)	34.0	2.12	2620

Very obviously, the applied stresses in the rings are several orders of magnitude below the critical stresses, and therefore acceptable.

#### Mass of rings

Finally, we can roughly compute the masses of the rings, assuming that they are less in mass than a cylinder of similar radius and thickness.

$$M = (\text{density})(\text{volume}) < (\rho)(\pi)(D)(t)(h)$$

where for each ring

$$\rho = 1826 \text{ kg/m}^3$$

$$D = 35.0 \text{ cm}$$

$$t = 5.0 \text{ mm}$$

$$h_{upper} = 9.0 \text{ cm}$$

$$h_{lower} = 15.0 \text{ cm}$$

---


$$\text{Mass of upper launch ring} = 0.90 \text{ kg}$$

$$\text{Mass of lower launch ring} = 1.50 \text{ kg}$$


---

## **4 Summary**

This MSP structural design, then, represents a involved attempt to satisfy mission criteria for a spacecraft designed to return of a sample of Martian soil for Earthbound analysis.

Although at points overdesigned, and in some areas not detailed enough, the structure (or at least its critical components) have been shown to be able to withstand some of the major loads placed upon them by the harsh and varied environments the craft would encounter on its round trip to and from the Red Planet.

## References

---

1. A.C. Urugal & S.K. Fenster, "Advanced Strength and Applied Elasticity", Elsevier, N.Y., 1987
  2. E.F. Bruhn, "Analysis and Design of Flight Vehicle Structures", Jacobs, Indianapolis, 1973
- Various NASA Technical Reports:
3. R.E. Hutton, "Mars Surface Soil Erosion Study", CR-100822
  4. V.J. Blanchard, "Full-Scale Dynamic Landing Impact Investigation of Proto-type Lunar Module Landing Gear", TN-D-5029
  5. J.R. McGehee, "An Expandable Gas Bag Concept for a Stowable Omnidirectional Multiple-Impact Landing System", TM-X-59623
  6. J.L. McCarty, "Experimental Study of Vertical Impacts of a Lunar-Module Type Landing Gear Assembly Under Simulated Lunar Gravity", TN-D-4711
  7. O.R. Otto, "Analysis and Limited Evaluation of Payload and Legged Landing System Structures for the Survivable Soft Landing of Instrument Payloads", CR-111919

## MSP Structural Designer Quarterly Time Sheet

Date	Time	Activity
4/6	2.25* hr	Discussion of layout and propulsion basics, mission scenario
4/7	0.50	Research on Viking, computer sketches
4/8	1.00	Preliminary MSP sketches
	1.50	
	2.25	
4/12	2.00*	Assessment of MSP I, sketches for MSP II Computer drawings of MSP I, nozzle detail Finalized layout; components, shield, chute, gear
4/23	1.00*	Discussion with Tim on component layout
	4.00	Layout of fuel, structure, fuel systems on computer
5/6	4.00	Research on landing gear
	1.50	Sketches of gear, calculations
5/13	2.50	Finalizing layout sketches
	1.00	Initial shock absorber calculations
5/16	1.50	More shock calc's
	3.00	Landing gear plan views, load calc's
5/20	2.00	Final general layout details
5/24	2.50*	Decisions made on tasks finalizations of tasks, synthesis
5/28	3.00	Stress calc's on bus, gear, platform
5/29	2.00	MSP final drawings started
5/30	1.00	First text for final report written
5/31	6.00	Final CAD work on MSP components
6/1	4.00	Text writing for final report
	2.00	FirstLaser printing of report
	1.00	Pasted report
	6.00	Finished text, drawings of report
6/2	2.00	Laser printed report
	2.00	Pasted, photocopied report
<b>TOTAL</b>	<b>61.50+</b>	<b>Hours for quarter</b>

\*Involved group meetings (some not listed)

Section B:  
**Thermal Control and Guidance**

Terrance Yee

## SPACECRAFT THERMAL CONTROL

Thermal design for this craft is primarily driven by the need to keep the propellant tanks between +10 and +55 degrees celcius and the batteries between 0 and +25 degrees celcius.(ref D6 ) The most critical conditions are during nighttime on the surface and during ascent in the atmosphere. To remain within the allowable temperature range a variety of suface coatings, insulation, and active cooling and heating units are employed.

### Orbital Thermal Control

Thermal control before descent is accomplished though proper surface coatings and orientation of the spacecraft. After release from the MOV and a short burn to precess in its orbit the MSP assumes a slow "rotissary" mode of rotation about its axis of symmetry. The bottom of the hexagonal bus section is gold coated ; the top of the hexagonal bus is painted white and the sides are vertically striped with 50% gold and 50% white paint. This paint scheme together with the shadow of the heat shield allow the spacecraft to alter the effective absorbtivity vs. emissivity ratio by changing the spacecraft angle with respect to the sun. The angle of the symmetry axis with respect to the incident radiation is chosen by the onboard computer to give a daylight (non-eclipse) equilibrium temperature of 20C. If the spacecraft begins to warm the computer will tilt the top of the craft towards the sun thereby shadowing some of the craft with the heat shield and exposing the white top of the hexagonal bus. This manuever will also be used prior to reentry to precool the craft to 15C which will lessen the impact of the aerodynamic heating on reentry. If during orbit the spacecraft is too cool the top can be tilted away from the sun exposing the gold bottom of the hexagonal bus which will tend to warm the craft.

This flexible effective surface was chosen due to the eccentricity of the martian solar orbit which alters the intensity of solar radiation depending on what time of year the mission is

performed during. It also allows us to be less precise in our calculation of the required mix of coatings. Those calculations were performed by analyzing the steady state radiation equilibrium setting the incident solar radiation and albedo radiation equal to the thermal radiation leaving the craft. Approximating the craft by a sphere of radius  $R$  the following results were obtained for the different mars-sun distances:

$$Q_{\text{em},i} = Q_{\text{abs}} \quad \epsilon 4\pi R^2 G T^4 = \alpha_s (I_s + I_{\text{alb}}) \pi R^2 \quad I_{\text{alb}} = \frac{\alpha_s}{\epsilon} \left[ 1 - \sqrt{1 - \frac{R_m}{R_m + 600\text{km}}} \right]$$

$$T = 293.15^\circ\text{K}$$

$$\text{minimum mars-sun distance} \rightarrow \frac{\alpha_s}{\epsilon} = 2.356$$

$$\text{maximum mars-sun distance} \rightarrow \frac{\alpha_s}{\epsilon} = 3.429$$

selecting gold ( $\frac{\alpha_s}{\epsilon} = 6$ ) and Titanium Oxide white paint with Methyl (.222)

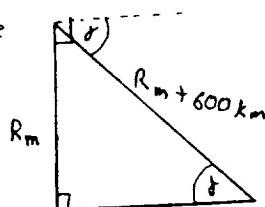
$\Rightarrow$  minimum distance : 37% gold, 63% white paint

maximum distance : 56% gold, 44% white paint

For an eclipse the following relations were used to find the time of the eclipse:

for a 600 km altitude orbit above mars the angle that the penumbra makes with respect to the mars-sun radius is only  $0.193^\circ$  so we can approximate this angle as  $0^\circ$ .

therefore we can write



$$\Rightarrow \gamma = 58.22^\circ$$

$$T = 2\pi \mu^{1/2} (R_m + 600\text{km})^{3/2}$$

$$t_{\text{ecl.}} = \frac{2\gamma}{360^\circ} T$$

$$\rightarrow t_{\text{ecl.}} = 2486\text{s} = 41\text{min } 26\text{sec}$$

Using a lumped capacitance approximation for spheres of different radii the change in internal thermal energy is set equal to the energy radiated away during the eclipse. Specifying the allowable change in temperature as ten degrees celcius allows us to solve for the surface temperature. Once this is known a minimum ratio of insulation thickness to thermal conductivity can be derived from fourier's law of conduction. The actual thickness of insulation is seen to be greater than the minimum required so the craft will stay within the allowable temperature range. An analysis was also performed assuming no insulation and use of electric resistance heaters during eclipse as a comparison.

$$\text{assuming } C_p \approx 1.0 \frac{\text{kJ}}{\text{kg K}}$$

$$m C_p \Delta T = t_{\text{ecl}} \epsilon 4\pi R^2 G T_s^4 = 4\pi R^2 \frac{K}{L} (T - T_s) L_{\text{cpl}}$$

$$\text{if no insulation } P = 4\pi \epsilon R^2 G T_s^4$$

$$\text{assuming } T_s = 293 \text{ K}$$

$$\rightarrow \frac{K}{L} = 0.334 \frac{W}{m^2 K} \quad (\text{for } R = 1 \text{ m})$$

$$\text{actual } \frac{K}{L} \text{ used} = \frac{2.9 \times 10^{-3}}{.01} = .29 \frac{W}{m^2 K}$$

1cm Refrusil (see ref D1 pg 131)

R(m)	P required (W)
.5	612.1
.75	1377.2
1.0	2448.5
1.5	5509.03

## Reentry Heating

The allowable temperature change on reentry is specified as 5 degrees Celcius which allows a factor of saftey of 2 before the batteries overheat. Originally an ablative heat shield was envisioned but after considerable research and a more accurate determination of the reentry heating it was found that a radiative heat shield using a high temperature multilayer insulation would function quite well in this environment.

The reentry heating of the shield was approximated by calculating the stagnation enthalpy and pressure at the stagnation point for numerous points during the descent then the heat transfer

rate was found using the chart provided. Setting the total heat transfer rate to the shield equal to the heat radiated away from the front surface allowed us to solve for the maximum temperature at this point on the front of the shield. This allowed us to choose the type and thickness of insulation to give the desired heat transfer and backface temperature. The beryllium honeycomb on the backface was not allowed to exceed 700 degrees Fahrenheit. It was also discovered that the thermal lag for a heat shield of about 45 seconds allowed for an additional factor of safety in the backface support structure since it will be at a lower temperature than equilibrium when it experiences the maximum drag force.(ref )

Two computer programs were written for this section. The first helped collate the atmospheric, trajectory, and heating rate tables and charts. The second calculated a few stagnation temperatures outside the boundary layer in front of the shield assuming no dissociation (not a very good assumption) in order to get a rough idea of how much error is induced by assuming a 300 degree Kelvin wall in using the given handout since a very high freestream temperature would make the difference in heat shield temperature less important.(see appendix D1)

$$\dot{q}_{\max} = \varepsilon G T_{s\max}^4 \rightarrow T_s = 991.3K$$

$$T_{back} = 644K$$

assuming a view factor of  $\frac{1}{4}$   
between the craft and the back of the shield  
 $\rightarrow \dot{q}_{allow} = 11,111 \text{ W for 180 seconds}$   
(heat shield)

$L_{reg} = 0.112 \text{ mm of Aluminum fiberglass multilayer blanket (max temp = } 1033K)$   
minimum thickness manufactured = 5mm,  $f = 181b/f_3$

so our 5mm thick insulation weighs 2.55 kg and keeps the back face very cool with almost no heat transfer to the vehicle.

In the recirculation zone behind the heat shield the convective heat transfer is approximately one tenth of that experienced at the stagnation point.(ref D3 ) Approximating the temperature of the recirculation zone as 1400K and modeling the craft as a cylinder we get the following convective heat transfer relations for the bus during descent:

$$h_{\text{stagn. pt.}} = \frac{\dot{q}}{\Delta T_{\text{chart}}} = \frac{46500 \frac{W}{m^2}}{6000 K} = 7.75 \frac{W}{m^2 K}$$

$$h_{\text{stagn.}} = 10^6 h_{\text{stagn. pt.}} = 0.775 \frac{W}{m^2 K}$$

5

(ref D9)

[approximate  $r = \frac{L}{2}$ ,  $L = 1 m$ ]

$$mc_p \Delta T_{\text{craft}} = Q = 180 \text{ sec} \cdot \left[ \frac{1}{7.75 \frac{W}{m^2 K} \cdot 2 m^2} + \frac{\ln r_0/r_i}{2 \pi K_{\text{inst.}}} \right]^{-1} \Delta T_s$$

$\sim 100 K$

allowing  $\Delta T_{\text{craft}}$  of  $5 K$   $\rightarrow r_0 - r_i = .7 \text{ mm}$  Refrasi which is well below the  $1 \text{ cm}$  of Refrasi actually on the craft.  
the actual  $\Delta T_{\text{craft}} \approx 0.4^\circ K$  with  $1 \text{ cm}$  Refrasi

THIS PAGE IS  
OF PUBLIC DOMAIN

In order to maintain stability during descent a reinforced carbon-carbon "drogue chute" is deployed when entering the atmosphere. This cone-shaped body is tethered to the rear of the craft on either side of the motor by high temperature titanium cables. The drag cone is stored as the lid to the main chute canister and when deployed by coil spring will trail 10 meters behind the craft. Since the heat shield stagnation point was at  $991 K$  we assumed this temperature as the worst case for the entire cable. To size the drag cone we assumed a  $50 \text{ m/s}$  gust which is approximately a 2% variation in drag force (about  $100 \text{ N}$ ). Letting this disturbance force act at  $75 \text{ cm}$  from the center of the heatshield leads to a  $75 \text{ Nm}$  disturbance torque. If we allow a maximum moment arm of the drag cone force of  $1 \text{ m}$  then we require a minimum force of  $75 \text{ N}$  which translates into an area of  $220 \text{ cm}^2$  if we assume that the drag force per area of the heatshield is equal to the drag force per area of the drag cone. With a right cone ( $45$  degree half angle) this translates into a surface area of  $707 \text{ cm}^2$  and a base radius of  $15 \text{ cm}$ . Adding a factor of safety to this, we increase the base radius to  $21 \text{ cm}$  and the surface area to  $980 \text{ cm}^2$ . Making the cone out of  $5 \text{ mm}$  reinforced carbon-carbon (RCC) leads to a cone weight of  $784 \text{ g}$ . A simple stress analysis of the cable shows that a diameter

of 3.175mm is capable of supporting more than 1000N force and weighs 360g. Therefore a total weight estimate of the stability system is 1.3kg allowing 156g for fixtures such as bolts and a spring.

## Thermal Control on the Surface

After reentry the MSP may be warmer than desired, therefore a method of rapid cooling is required. The method chosen to accomplish this is three small louvers which are placed 120 degrees apart on the ascent body above the ascent fuel tanks. The louvers are opened by thermally set springs and held shut by latches which the computer opens upon landing and locks at night. They are top hinged, opening outward to expose an 8cm high, 3cm wide opening for the martian atmosphere to flow through.

Some concern was originally felt for the cooling effect of the expanding helium in the propulsion system. The expansion is solved using the ideal gas law:

$$\frac{P_1 V_1}{P_2 V_2} = \frac{T_1}{T_2}$$

$$T_1 = 293K \quad V_1 = 4.905 \times 10^{-3} m^3$$

$$P_1 = 20 \text{ MPa} \quad V_2 = 2.9575 \times 10^{-2} m^3$$

$$P_2 = 3 \text{ MPa}$$

$$\rightarrow T_2 = 220.8K$$

$$m C_p \Delta T = 75 \text{ kJ} \text{ for a } .2 \text{ kg He tank}$$

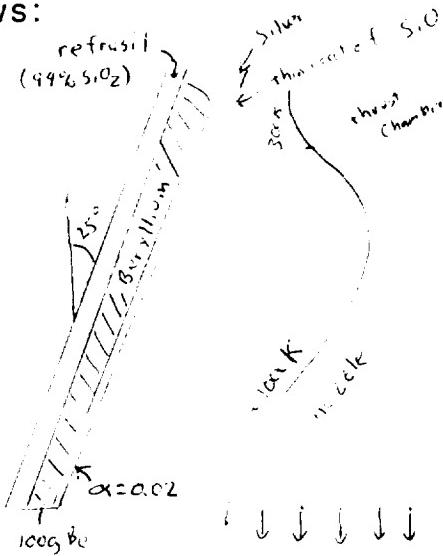
$$\rightarrow \Delta T_{\text{MSP}} = -0.75^\circ\text{C per He tank}$$

during descent

$$\Delta T_{\text{MSP}} = -1.5^\circ\text{C per He tank during ascent}$$

This temperature decrease is certainly acceptable during the rocket firings while heat from the rocket motor and aerodynamic heating are offsetting the helium expansion effects.

This brings us to the next problem area, the rocket motor. While the thrust chamber is ablatively cooled and has negligible heat transfer to the rest of the ship, the nozzle and throat are radiatively cooled and must be isolated from the rest of the ship. This is done using a three layered system consisting of a highly reflective outer coating of silver with a protective coating of SiO (150 Angstroms), a layer of beryllium as a heat sink, and a final layer of refrasil, a multilayer high temperature insulation. The analysis of this system follows:



A worst case approximation of the system would be as parallel plates

for which

$$q_{12} = \frac{\epsilon_1 \epsilon_2 G (T_1^4 - T_2^4)}{\epsilon_1 + \epsilon_2 - \epsilon_1 \epsilon_2}$$

(ref. 89)

$$(T_2 = 293), 1 \text{ kg} \times 1825 \frac{\text{J}}{\text{kgK}} = q_{12} \tau_{burn}^{62 \text{ sec. max}}$$

$$\rightarrow T_2 = 622 \text{ K}, q_{12} \tau_{burn} = 61 \text{ kJ}$$

$$\text{thermal resistance to conduction} = 3.45 \frac{\text{m}^2 \text{K}}{\text{W}} \quad \text{cooling due to radiation} = \frac{1}{4} G \epsilon T^4$$

$$q_{load} = \frac{329}{3.45} = 95.4$$

view factor

$$q_{load} = 21.2 \frac{\text{W}}{\text{m}^2} @ 622 \text{ K}$$

$\rightarrow \frac{4}{5}$  of energy transferred to spacecraft

$\rightarrow 48.9 \text{ kJ}$  to craft

$\rightarrow 2.4^\circ\text{C}$   $\Delta T$  for ascent stage

$\rightarrow +5^\circ\text{C}$  for deorbit burn

The last ground problem is cooling at night due to radiation loss and convection. It is this area which requires the most insulation and indeed sizes the insulation used. A small UDMH decomposition fuel cell is used to warm the ship at night.

$$\text{since pressure stays the same at day & night} \quad \rho_{night} = \rho_{day} \frac{T_{day}}{T_{night}} = 1.96 \times 10^{-5} \frac{\text{g}}{\text{cm}^3}$$

$$M = 91.5 \times 10^7 \frac{\text{kg}}{\text{ms}} \quad \rightarrow V = 4.67 \times 10^{-4}, \quad K_{180\text{K}} = .00828 \frac{\text{W}}{\text{mK}}$$

$$C_{P,180\text{K}} = 770.4 \frac{\text{m}^2}{\text{s}^2 \text{K}} \quad \rightarrow a = 5.484 \times 10^4 \frac{\text{m}^2}{\text{s}} = \frac{\text{K}}{C_P \cdot \rho}$$

$$Pr = \frac{\gamma}{\eta} = 0.852 \quad Re = \frac{V \cdot \rho}{\nu} = \frac{20 \text{ m/s} \cdot 0.34 \text{ m}}{4.67 \times 10^{-4} \frac{\text{m}^2}{\text{s}}} = 14561$$

using the relation

$$Nu = K Re^m Pr^{1/3} \quad w/ \quad K = .195, m = .618 \quad (\text{ref 90})$$

for  $Re = 14561$

$$Nu = 69.1$$

$$h = Nu \frac{K}{\ell_0} = 1.68 \frac{\text{W}}{\text{Km}^2}$$

Convection  
or heat transfer is  
dominant

$$q = \frac{293 - 180}{\frac{1}{2\pi \cdot 17 \text{ m} \cdot 1.68 \frac{\text{W}}{\text{Km}^2}} + \frac{\ln(\frac{18}{17})}{2\pi \cdot 2.4 \times 10^{-3} \frac{\text{W}}{\text{mK}}} \underbrace{- Y}_{1 \text{ cm retransl.}} = 30.6 \text{ W}$$

over a uniform  $180^{\circ}$  night  $Q = 1.327 \times 10^6 \text{ J lost}$

9

given that the heat of combustion is  $46 \times 10^6 \text{ J/Kg}$  for Kerosene

we estimate that hydrazine has at worst  $\frac{1}{3}$  of this value  
since the actual value is unavailable.

This leads to 86 grams of Hydrazine required to  
maintain temperature levels.

## Ascent Heating

Using the same methods as the descent the ascent trajectory was analyzed but with a nose made of RCC and .96cm of refrasil protecting the skin. Again as in the descent, the craft was allowed to cool to 15C before launch in order to alleviate heating constraints.

## Rendevous Vehicle

The rendevous vehical has very little on board that is temperature sensitive; the electronics are very simple and can be built to withstand wide temperature variations, there are no batteries or fuel, only superstructure, dirt, solar cells and an omnidirectional radio beacon.

The surfaces not covered by solar cells are covered by gold to keep the temperature near 290K.

## GUIDANCE AND POWER SUPPLY

Guidance is provided by inertial rate gyros and by sun and star sensors when outside the atmosphere. Final landing guidance is provided by a system which uses a prescanned map of the landing area which is stored in on board ROM computer memory.(see appendix D2) The map records safe landing sites and local landmarks for the entire probable region of landing. Just after the parachute is jettisoned and before the retro manuever a snapshot image is taken of the terrain and matched by landmarks and distinct features to the map stored in memory. Once the computer determines where it is, it can use the map to direct the craft to the closest known safe landing zone. A laser range finder is used to determine distances to the ground to allow the computer to make the decisions when to engage the main chute, when to jettison the chute, and when to engage and disengage the retro motor.

The sun and star sensors along with inertial rate gyros and an accurate timer allow the ascent computer to determine the correct timing of the firing of the apogee kick motor as well as its correct orientation. Also the ascent computer will set a timer on the apogee motor that will pyrotechnically pierce the combustion chamber to effectively kill the thrust when the correct orbit has been reached.

The on board computer actually consists of three parts. The first part is the descent computer which is the ROM board and the guidance for descent. This part is located in the hexagonal bus section and weighs about 1kg. The second part is the main or ascent computer. It is located in the ascent stage and weighs just under 2kg. Its functions include powered flight guidance, directing the sample collector, and orbital equations. The third part is a subpart of the main computer. Its function is to periodically check all systems, especially thermal, during the dormant periods while the MSP waits for a particular orbital conjunction. It has limited control over the attitude jets and thermal system and will activate the main computer if a serious problem is detected. It has negligible power usage and negligible mass.

Power is supplied to all but the rendevous stage by zinc-silver oxide batteries with a power density of 80 watt-hr/lb.(ref D2) There are two groups of batteries, one in the ascent stage and one in the hexagonal bus. The batteries for the sample collection motor are subdivided into many smaller cells in order to give a high voltage while the other batteries supply a lower voltage usable by the computers. The main computer system is assumed to operate at 30watts and the descent computer at 8 watts, the monitor subcomputer is estimated to have such a low power usage as to be negligible.

action	power(W)	time(min)	energy(watt-hr)
ascent	30	35	17.6
sample collection	330	10	55
final descent	38	5	3.17
deorbit and reentry	30	60	30
adjusting precession	30	40	20

burn to precession	30	10	5
unforeseen/reserves			27.63
total	158.4		

Total battery weight is 0.9kg, 0.8kg is stored in the hexagonal bus.

The beacon on the rendevous stage is an omnidirectional beacon broadcasting a narrow band 100W signal of 1 millisecond duration five times every second. To provide this power 145cm<sup>2</sup> cross sectional area of solar cells must be used. On the 10cm diameter rendevous stage this translates into a 14.5cm strip around the cylinder of 11% efficient silicon cells at furthest distance from the sun. If 17% efficient GaAs cells are used the strip will be 9.4 cm wide. The remainder of the surface should be covered in gold to keep the craft warm. Silicon solar cells are currently available and their reliability is high so they were chosen for this mission, giving a weight of 0.15kg to the power system.

## REFERENCES

- D1. NASA SP-5027, "Thermal Insulation Systems" ,Technology Utilization Publication, Cambridge, Massachusetts,1967
- D2. N63-11511,"Power for Spacecraft",NASA University Conference on the Science and Technology of Space Exploration, Washington D.C. 1962
- D3. NASA SP-23,"Aerodynamics of Space Vehicles", NASA University Conference on the Science and Technology of Space Exploration, Washington D.C. 1962
- D4. John D. Anderson Jr., "Fundamentals of Aerodynamics",MC Graw-Hill, 1984
- D5. Frank Incropera and David De Witt, "Fundamentals of Heat and Mass Transfer",John Wiley and Sons, 1990
- D6. Brij N. Agrawal, "Design of Geosynchronous Spacecraft",Prentice-Hall Inc., 1986
- D7. AGARD-CP-411, "Advances in Guidance and Control Systems and Technology", NATO 1986
- D8. Pruser,Faget,Smith, "Manned Spacecraft: Engineering Design and Operation", Fairchild Publications,1964
- D9. Meyer, R. X. , various class handouts, Winter and Spring 1990,  
UCLA

MSP THERMAL/POWER/GUIDANCE/COORDINATOR TIME SHEET

WEEK	HOURS
1	3
2	5
3	3.5
4	5
5	8
6	10
7	6
8	15
9	20
10	25
total	100.5

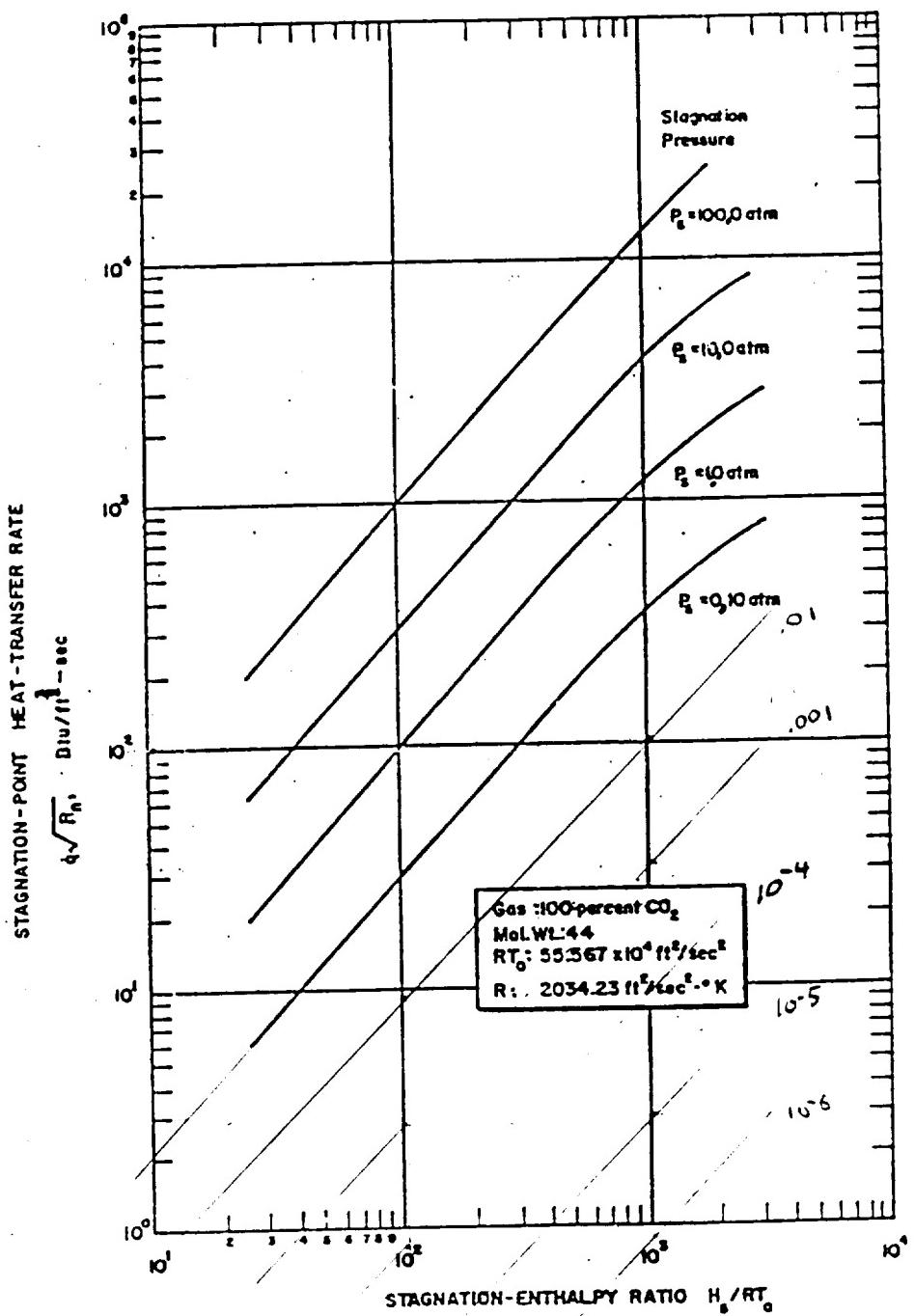


Figure 1 CONVECTIVE HEATING AT A STAGNATION POINT  
(100-PERCENT CO<sub>2</sub>)  
63-10199

For wall temperature = 300°K.  
chemical reactions are included (multi-component diffusion is neglected)

$\dot{q}$  = heat transferred to wall, per unit area and unit time.

$R_n$  = nose radius

$H_s$  = stagnation enthalpy

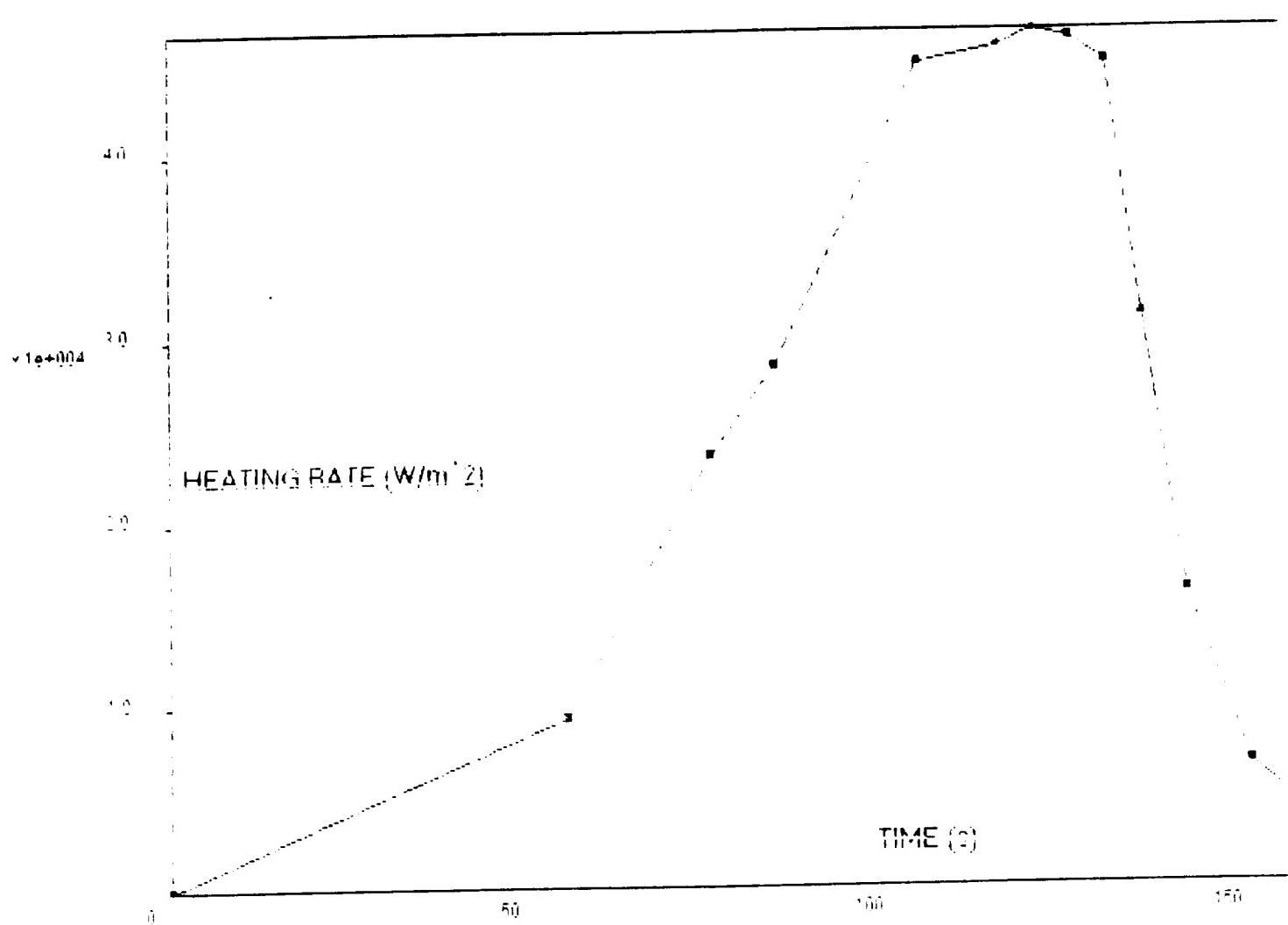
$$\dot{q} = n_w \frac{H_s - T_w}{c_p T_w}$$

### III. BOUNDARY-LAYER SOLUTIONS

Heat-transfer rates and boundary-layer profiles have been found from a similar solution of the laminar boundary-layer equations. Local similarity assumptions are used to reduce the equations to a set of five first-order total differential equations with variable coefficients. The solution is then found by an iterative technique using 400-point profiles. A detailed description of the solution is also given elsewhere.<sup>2</sup>

A number of assumptions are involved in this boundary-layer solution and should be mentioned for completeness: The atomic composition is assumed to be constant throughout the boundary layer. In essence, this means that multicomponent diffusion is neglected. The local similarity assumptions are exact for the case studied in this report. All fluid properties are continuously variable throughout the boundary layer. The effects of chemical reactions are included by employing the reaction-conductivity concept.

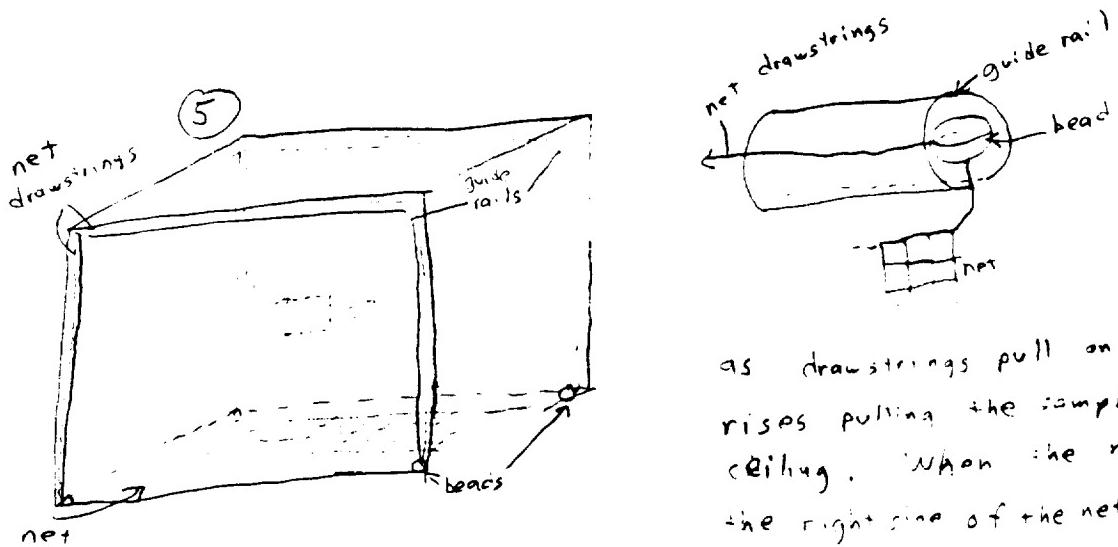
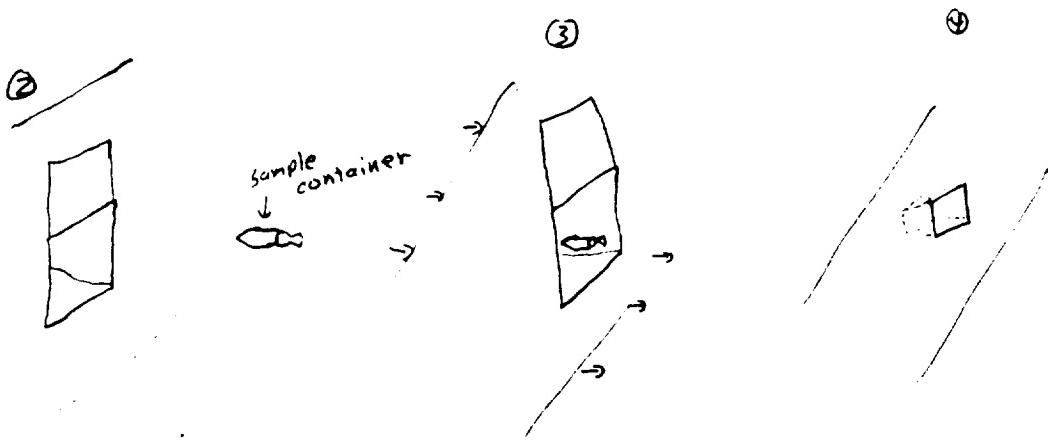
All calculations in the report have assumed a wall temperature of 300°K. Similarly, only stagnation-point solutions are reported here. Other cases with more realistic wall temperatures and other pressure gradients can be obtained by application of this method.



1.2  
1.0  
0.8  
0.6  
0.4  
0.2  
0.0

## Capture of sample container

- ① MOV zeroes in on beacon, waits until closest approach, then maneuvers to docking distance.
- ② MOV opens door to bay which housed MSP
- ③ MOV maneuvers to capture sample container inside bay
- ④ MOV closes bay doors, sealing container inside
- ⑤ net system inside bay draws container tight against a corner  
read for return trip to earth. see diagrams:



as drawstrings pull on beads the net rises pulling the sample toward the ceiling. When the net reaches the top the right side of the net is reeled along the ceiling while the left side is reeled through a gap at the top left edge. this process continues until the sample is held firmly against the upper left edge of the docking bay

STAGNATION PRESSURE ON SHIELD IS 9.324668E-07

X AXIS IS 112.718

HEATING RATE (W/m<sup>2</sup>)= 88.66328

VELOCITY(m/s)= 3520

ALTITUDE(km)= 150

STAGNATION PRESSURE ON SHIELD IS 1.227144E-04

X AXIS IS 231.8017

HEATING RATE (W/m<sup>2</sup>)= 9809.644

VELOCITY(m/s)= 3561

ALTITUDE(km)= 100

STAGNATION PRESSURE ON SHIELD IS 1.155288E-03

X AXIS IS 203.1755

HEATING RATE (W/m<sup>2</sup>)= 13495.77

VELOCITY(m/s)= 3551

ALTITUDE(km)= 80

STAGNATION PRESSURE ON SHIELD IS 3.14366E-03

X AXIS IS 188.1773

HEATING RATE (W/m<sup>2</sup>)= 18371.25

VELOCITY(m/s)= 3520

ALTITUDE(km)= 70

STAGNATION PRESSURE ON SHIELD IS 7.704226E-03

X AXIS IS 166.9835

HEATING RATE (W/m<sup>2</sup>)= 44774.96

VELOCITY(m/s)= 3425

ALTITUDE(km)= 60

STAGNATION PRESSURE ON SHIELD IS 1.679816E-02

X AXIS IS 139.1813

HEATING RATE (W/m<sup>2</sup>)= 45661.6

VELOCITY(m/s)= 3220

ALTITUDE(km)= 50

OPTIONAL  
OR FASTER QUALITY

STAGNATION PRESSURE ON SHIELD IS 2.303351E-02

X AXIS IS 120.0036

HEATING RATE (W/m<sup>2</sup>)= 46548.23

VELOCITY(m/s)= 3020

ALTITUDE(km)= 45

STAGNATION PRESSURE ON SHIELD IS 2.919942E-02

X AXIS IS 97.29362

HEATING RATE (W/m<sup>2</sup>)= 46104.9

VELOCITY(m/s)= 2750

ALTITUDE(km)= 40

STAGNATION PRESSURE ON SHIELD IS 3.438264E-02

X AXIS IS 74.39153

HEATING RATE (W/m<sup>2</sup>)= 44774.96

VELOCITY(m/s)= 2400

ALTITUDE(km)= 35

STAGNATION PRESSURE ON SHIELD IS 3.505111E-02

X AXIS IS 49.95343  
VELOCITY(m/s) = 1950  
ALTITUDE(km) = 30

HEATING RATE (W/m<sup>2</sup>) = 31032.15

STAGNATION PRESSURE ON SHIELD IS 3.192682E-02

X AXIS IS 30.8775  
VELOCITY(m/s) = 1500  
ALTITUDE(km) = 25

HEATING RATE (W/m<sup>2</sup>) = 15959.39

~~incorrect data~~ STAGNATION PRESSURE ON SHIELD IS 8.349418E-02

X AXIS IS 49.79182  
VELOCITY(m/s) = 1990  
ALTITUDE(km) = 20

HEATING RATE (W/m<sup>2</sup>) = 0 ALTITUDE(km) = 20

STAGNATION PRESSURE ON SHIELD IS 2.101189E-02

X AXIS IS 15.34466  
VELOCITY(m/s) = 990  
ALTITUDE(km) = 20

HEATING RATE (W/m<sup>2</sup>) = 6649.746

STAGNATION PRESSURE ON SHIELD IS 1.179943E-02  
X AXIS IS 8.059934  
VELOCITY(m/s) = 600  
ALTITUDE(km) = 15

HEATING RATE (W/m<sup>2</sup>) = 3103.215

31032.15  
15959.39  
0  
6649.746  
3103.215

```

10 CLS
20 REM*****PROGRAM TO DO HEAT SHIELD HEATING (HARD)*****
30 COLOR 7
40 INPUT "ALTITUDE (km)":ALT
50 INPUT "STATIC TEMP AT INFINITY(K)":T1
60 COLOR 9
70 INPUT "STATIC PRESS AT INFINITY(mb)":F1
80 COLOR 10
90 INPUT "SOUND SPEED AT INFINITY(m/s)":A
100 COLOR 11
110 MW=43.824
120 COLOR 12
130 INPUT "VELOCITY (m/s)":V
140 COLOR 14
150 PRINT
160 GAMMA=A^2*MW/(8314*T1)
170 CP=8314/MW*GAMMA/(GAMMA-1)
175 PRINT "CP":CP."830"
180 HS=CP*T1+V^2/2
190 X=HS/8314*MW/T1
200 M=V/A
201 PRINT "GAMMA":GAMMA
204 PRINT "M":M
210 P2=P1*((GAMMA+1)^2*M^2/(4*GAMMA*M^2-2*(GAMMA-1)))/(GAMMA/(GAMMA-1))
220 P2=P2*(1-GAMMA+2*GAMMA*M^2)/(GAMMA+1)/1013.25
230 PRINT"STAGNATION PRESSURE ON SHIELD IS":P2
235 LPRINT"STAGNATION PRESSURE ON SHIELD IS":P2
240 PRINT"X AXIS IS":X
245 LPRINT"X AXIS IS":X
250 COLOR 13
260 INPUT "INPUT Q*SQR(Rn) IN Btu/(Ft^(3/2)*sec)":Q1
270 COLOR 15
280 Q1=Q1/SQR(2*100/2.54/12)/9.4787E-04/144/2.54^2*100000
290 PRINT"HEATING RATE IN W/m^2 IS":Q1
300 PRINT
310 PRINT
320 LPRINT "VELOCITY(m/s) = ":V, "HEATING RATE (W/m^2) = ":Q1, "ALTITUDE(km) = ":ALT
330 LPRINT
340 GOTO 30

```

AMMA	VELOCITY	SOND SPEED	STATIC TEMP OF STREAM	STAB. TEMP
1.106	3185	184	148	5006.858
1.14	3231	193	168	5994.14
1.3	3194	217	190	6364.408
1.345	3083	217	201	6546.136
1.34	1813	233	213.5	5011.551
1.33	2133	235	216	5633.931
1.324	1420	240	228	1821.016
1.322	522	245	241	417.1575
1.323	1000	244	236.2	876.9264
1.32	429	247	242.5	359.5449

OPPOSITE  
OF PULSED QUALITY

Section C:  
**Propulsion Systems**

Myles Baker

## Propulsion:

### 1: Requirements:

The Mars Surface Probe places several demands on its propulsion system. From the Mission Scenario, at least five separate impulses are required:

Event:	Required Delta V:	Required Thrust:
Initial orbital maneuvering	<10 m/s	< 10000 N
Descent burn	<200 m/s	< 10000 N
Landing retro burn	<200 m/s	> 500 N
Ascent from Mars surface	~3920 m/s	> 1000 N
Apoapsis injection	~830 m/s	< 200 N

In addition, the propulsion system should be as small and light as possible, since mass is at a premium in such a mission. It should be as simple as possible to increase reliability, and it should be very stable due to the approximately two years it will spend in flight to Mars. An additional restriction placed on the propulsion system is that it never cause the vehicle to accelerate at more than  $100 \text{ m/s}^2$ , which is reflected in the maximum allowable thrusts.

With these criteria in mind, the decision was to use two separate engines; one for the low-thrust apoapsis injection burn and the other for all other, higher thrust burns. Due to the varying requirements of the first four impulses, it was decided to use a throttleable, re-startable liquid propellant engine. On the other hand, because of the decision to have no active guidance on the final payload capsule, the advantages of a solid motor far outweigh its slightly lower specific impulse. These

advantages include the lack of propellant control devices (valves, etc.), propellant tanks, and the fact that the pre-determined impulse delivered by a solid motor does not require any outside control mechanisms (other than a simple ignitor).

---

#### Liquid Propellant Engine:

The prime consideration in the design of this engine was its simplicity. There are several choices in the design of a rocket engine, each of which has a great impact on the final design of the vehicle. Some of the options, and selections for this mission, are listed below.

**Thrust:** The primary restrictions placed on the thrust of the engine are the imposed design restriction of less than 100 m/s<sup>2</sup> maximum acceleration and the need to overcome Mars' gravity with minimum gravity losses, which requires as high a thrust as possible. From Newton's second law of motion,

$$F_{t,\max} = M_{\min} a_{\max}$$

With a burn-out ascent mass of 15 kg, a thrust of approximately 1500 Newtons is required.

**Cooling:** There are several methods of cooling a rocket engine, ranging from simple radiation cooling to regenerative cooling by pumping cool fuel through a cooling jacket around the engine. Since this engine is rather small, the extra hardware and complexity required for regenerative cooling eliminate it as a viable option, while radiation cooling is very well suited for such small engines. However, due to the fact that the actual combustion chamber is located inside the vehicle during much of

the mission, some additional cooling is required. Since there is a relatively short total burn time for the engine (approximately 120 seconds), an ablative combustion chamber liner will be used. Such materials are most reliable at chamber pressures of 1 MPa or lower, so 1 MPa is chosen as the maximum chamber pressure for the MSP liquid engine.

**Fuels:** There are many available fuels for liquid rockets, and various parameters for selecting a fuel for a particular mission. In this case, our requirements are a high specific impulse, high density, and easy storeability for up to two years without maintenance. The latter requirement eliminates several of the higher-energy propellants, such as the cryogenics (Liquid Hydrogen) and the fluoridated oxidizers. On this mission it was decided to use Nitrogen Tetroxide as an oxidizer and 50% Unsymmetrical Dimethyl Hydrazine, 50% Hydrazine (50-UDMH) as a fuel. This propellant combination has several advantages, including hypergolic ignition (no separate ignition system is required), extremely high storeability, a relatively high specific impulse, and a broad base of engineering experience with such propellant combinations.

**Propellant feed:** There are two types of propellant feed systems in use today: pump-fed and pressure-fed. The pump-fed system requires some additional machinery (a turbo pump, gas generator, and gas turbine), but it allows lower propellant tank pressures, and hence thinner tank walls, lowering the total mass of a large vehicle. Pressure-fed systems, on the other hand, maintain the entire propellant supply at a high pressure and thus

require heavier tanks, but virtually no machinery other than valves. For a vehicle as small as the Mars Surface Probe, the slight increase in tank mass is greatly compensated by the lack of supporting machinery, so a pressure-fed system is more appropriate.

**Materials:** Recent advances in high-temperature, relatively oxidation-resistant composite materials makes the choice of materials for the motor structure fairly straightforward. Multi-cycle burn times well over the required 100 seconds have been demonstrated with engines constructed almost entirely from a 3-Directional Novoltex<sup>(R)</sup> type carbon fiber reinforcement in a silicon-carbide matrix<sup>[C-1]</sup>. In the previously built engines, the fuels, flame temperatures, and motor sizes were similar to those in the Mars Surface Probe.

---

**Solid Motor:**

The solid rocket motor for re-insertion into the Mars Orbiting Vehicle's orbit is much simpler. It is only required to ignite, deliver a pre-determined impulse, and extinguish. The payload capsule will already be positioned in the correct attitude and spin-stabilized by the ascent stage, so no attitude control will be necessary. With spin-stabilization, if the spin rate is high enough, it is not necessary to adjust the thrust vector assuming a small uncertainty, so the engine mechanics can be extremely simple. The spin-stabilization of the upper stage will be provided by unwinding a pre-stressed "clockspring" mechanism, in effect spinning the ascent and final stages against

one another.

The fuel used in the solid motor is also well-tested, off the shelf technology. It consists of Aluminum and Ammonium Perchlorate powders in a polyurethane binder. This compound was chosen primarily for its slow burning rate of 6 mm/s, which makes it possible to remain within the 100 m/s<sup>2</sup> design acceleration limit. To minimize the motor case mass and surface area, a spherical case will be used, with a nozzle very similar in its proportions to that used in the liquid engine. The materials used are also very similar to those used in the liquid engine, with the addition of a 4-Directional Carbon-Carbon Sepcarb<sup>(R)</sup> throat insert to minimize erosion in this critical area.

---

#### Propellant Tanks:

There are two necessary sets of propellant tanks. The first set is used for the orbital maneuvering, de-orbit, and retro burns and is housed in the main bus of the vehicle, which is left behind on the planet surface. It contains approximately 5 liters each of N<sub>2</sub>O<sub>4</sub> and 50-UDMH pressurized to 1.2 MPa, allowing for 5% ullage, and 2 liters of helium gas at 25 MPa to pressurize the tanks. These tanks will be split up into three identical sets of three tanks each, with similar tanks (e.g. N<sub>2</sub>O<sub>4</sub>, 50-UDMH, and He) arranged symmetrically about the axis of the vehicle. Due to the small size of the tanks, and the relatively low pressure, 2014-T6 Aluminum alloy (chosen for its good machining properties) will be used for the propellant tanks, and Ti 6Al-4V Titanium alloy (for its exceptionally high strength) for the Helium. Unfortunately,

Nitrogen Tetroxide is not compatible with aluminum tanks, so an interior protective coating will have to be used.

The second set of tanks, used only for the ascent burn, must contain about 19 liters of each of the propellants and 7 liters of helium, again allowing 5% ullage. The configuration for the ascent tanks is shown in figure C.7, with the materials similar to those in the descent tanks. The nested configuration was chosen in order to keep the ascent envelope as small as possible, and only weighs about fifty grams more than the optimum spherical tanks.

---

#### Propulsion System Analyses:

The design of a spacecraft is necessarily an iterative process, with the results from each part of the analysis affecting the parameters the remaining phases are based on. This necessitates a certain number of assumptions in the preliminary vehicle design, with successive refinements as the design progresses. The assumptions used in the analyses below have been through three such iterations, and so are fairly accurate.

---

#### Liquid Propellant Engine:

There are many parameters of interest in the design of a rocket engine, but perhaps the most important are the thrust ( $F_t$ ), specific impulse ( $I_{sp}$ ), mass ( $m$ ), and dimensions, which will be developed here.

The combustion characteristics of Nitrogen Tetroxide and 50-UDMH are tabulated in several references, each with slightly

different values. The values below are taken from Sutton and Ross [C-2]:

Flame Temperature:	3100 K
Gas Constant (R) of Exhaust:	396 J/(kg-K)
Ratio of Specific Heats ( $\gamma$ ):	1.24
Density: $N_2O_4$ :	1450 kg/m <sup>3</sup>
50-UDMH:	910 kg/m <sup>3</sup>
Mixture Ratio (O:F)	
by Mass:	1.62
by Volume:	1.01

Table X.X:  
Combustion Properties of Liquid Fuels

Nozzle: Assuming a ratio of  $\epsilon=20$  between the nozzle exit area ( $A_e$ ) and throat area ( $A_t$ ), the mach number (obtained through a computer iteration) at the exit plane is found to be  $M_e=3.93$ . Given the chamber pressure ( $P_c=1MPa$ ), the pressure at the nozzle exit plane ( $P_e$ ) can be obtained from the equations for isentropic flow:

$$P_e = P_c \times \left[1 + \frac{\gamma-1}{2} M_e^2\right]^{\gamma/(\gamma-1)}$$

$$P_e = 10^6 \times \left[1 + \frac{1.24-1}{2} (3.93)^2\right]^{1.24/(1.24-1)}$$

$$P_e = 4439 \text{ Pa}$$

Given the chamber pressure, the exit pressure, and the back pressure ( $P_b$ ), the thrust coefficient, defined as

$$F_t = P_c A_t C_f$$

can be obtained from the relation

$$C_{f, \text{ideal}} = \sqrt{\frac{2\gamma^2}{\gamma-1} \left(\frac{2}{\gamma+1}\right)^{\gamma/(\gamma-1)}} \left[1 - \left(\frac{P_e}{P_c}\right)^{(\gamma+1)/\gamma}\right] + \frac{P_e - P_b}{P_c} \epsilon$$

which, for this engine, at the mean surface pressure at the MSP landing site of 700 Pa, is

$$C_{f, \text{ideal}} = \sqrt{\frac{2(1.24)}{1.24-1} \left(\frac{2}{1.24+1}\right)^{\frac{1.24+1}{1.24-1}}} \left[1 - \left(\frac{4439}{1000000}\right)^{(1.24+1)/1.24}\right] + \frac{4439-700}{1000000} (20)$$

$$C_{f, \text{ideal}} = 1.77$$

Assuming a bell nozzle profile of maximum efficiency (a perfect bell nozzle of expansion ratio 30:1 truncated to 20:1), the nozzle losses (due to friction and nozzle divergence) are approximately 2%<sup>[C-2]</sup>, so the actual thrust coefficient is

$$C_f = C_{f, \text{ideal}} \times 0.98 = 1.73$$

Given a maximum thrust of 1500 N and a maximum  $P_c$  of 1 MPa, the required nozzle throat area is

$$A_t = F_t / (P_c C_f) = 1500 / (1.73 \times 10^6)$$

$$A_t = 9 \text{ cm}^2$$

For a throat area of 9 cm<sup>2</sup>, the inside diameter at the throat is 3.4 cm, the exit diameter is 15.1 cm, and from the bell nozzle optimization curves<sup>[C-3]</sup>, the nozzle length is 18.6 cm.

The mass flow through the nozzle can be obtained by a mass balance at the throat, where flow is necessarily sonic, so

$$\dot{m} = A_t P_c \sqrt{\frac{Y}{R T_c}} \left( \frac{2}{Y+1} \right)^{\frac{Y+1}{2(Y-1)}} \\ \dot{m} = (10^6 \text{ Pa}) (10^{-4} \text{ m}^2) \sqrt{\frac{1.24}{(386)(3100)}} \left( \frac{2}{1.24+1} \right)^{\frac{1.24+1}{2(1.24-1)}}$$

$$\dot{m} = 0.533 \text{ kg/s}$$

at a maximum thrust of

$$F_{t,\text{max}} = C_f A_t P_c = (1.73) (10^{-4}) (10^6)$$

$$F_{t,\text{max}} = 1560 \text{ N (at surface)} \\ 1575 \text{ N (in vacuum)}$$

This gives the liquid engine a specific impulse of

$$I_{sp} = \frac{F_t}{\rho \dot{m}}$$

$$I_{sp} = 298 \text{ sec (at surface)} \\ 301 \text{ sec (in vacuum)}$$

**Combustion Chamber Geometry:** In order to minimize pressure losses in the combustion chamber, the velocity just upstream of the throat should be very small, implying a large chamber to

throat area ratio ( $A_c/A_t$ ). In this case,  $A_c/A_t$  will be set to 6.0. Thus the radius of the combustion chamber is

$$r_c = \sqrt{6A_c/\pi}$$

$$r_c = \sqrt{0.0054/\pi} = 4.45 \text{ cm}$$

A second parameter in the design of a liquid-fuel combustion chamber is the characteristic length ( $l^*$ ), which is defined as the chamber volume divided by the throat area. This length is a useful measure of the amount of time a given element of the fuel-oxidizer mixture remains in the chamber, and must be high enough to ensure complete combustion. In the reaction control thrusters used on many early American launch vehicles, Monomethyl Hydrazine and  $N_2O_4$  were used as propellants and the characteristic length was approximately 0.5 meters<sup>[C-2]</sup>. This thrust chamber was used as a model, thus requiring a combustion chamber length (approximating the chamber as a cylinder) for the MSP engine of

$$l_c = l^*(A_t/A_c)$$

$$l_c = 0.5/6 = 8.33 \text{ cm}$$

Allowing for 5 cm clearance at the injector for fuel lines and the thrust vector control (gimballing) mechanism, the total liquid engine dimensions are:

$$\begin{aligned} D_{\max} &= 16 \text{ cm} \\ l_{\text{tot}} &= 30 \text{ cm} \end{aligned}$$

**Ablative Cooling Insert:** Ablative materials are often used in cooling small, short duration thrust chambers. When exposed to the heat of combustion, the ablator, a fiber reinforcement in an organic matrix, decomposes, leaving a porous, insulating layer of char on the surface. The decomposition of the organic matrix

is an endothermic process, and the pyrolysis gases released tend to insulate the remaining material from the hottest of the combustion chamber gases.

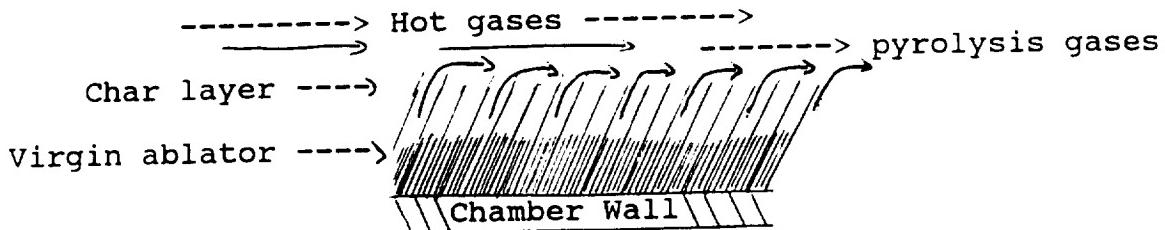


Figure C-1: Ablative Chamber Cooling

A silica fiber reinforcement in a phenolic matrix will be used as the ablative liner of the combustion chamber. The depth of the porous char layer, and hence the necessary thickness of the ablator, is given by the relation<sup>[C-3]</sup>

$$d_c = \sqrt{\text{burn time in seconds}} \text{ mm}$$

$$d_c = \sqrt{120} \text{ mm} = 11 \text{ mm}$$

This analysis does not consider the fact that the engine will reach ambient (spacecraft) temperature between firings, so the actual char depth should be significantly less. Nevertheless, adding a margin of safety, the thickness of the ablative material should be approximately 1.5 cm.

**Propellant Injectors:** Given the overall mass flow rate, the mixture mass ratio, and the propellant densities, it is easily seen that the necessary volumetric flows for maximum thrust are

$$Q_f = 0.225 \text{ l/s (50-UDMH)}$$

$$Q_o = 0.227 \text{ l/s (N}_2\text{O}_4)$$

and, given an orifice-type injector, the volumetric flow rate is

given by the equation<sup>[C-4]</sup>

$$Q = A_i C_i \sqrt{\frac{2\Delta P_i}{\rho}}$$

where  $C_i$  is an experimentally determined discharge coefficient, usually between 0.6 and 0.9 (assume 0.75), and  $P_i$  is the pressure drop across the injector. For a set of  $n$  injectors, the total area is

$$\sum_i^n A_i = \frac{Q}{C_i} \sqrt{\frac{\rho}{2\Delta P_i}}$$

which is, for a 200 kPa pressure drop,

$$A_{i,f} = 1.43 \times 10^{-5} \text{ m}^2$$

$$A_{i,o} = 1.82 \times 10^{-5} \text{ m}^2$$

For efficient combustion, it is desirable to have as many doublet-impinging pairs of injectors as possible in order to atomize the propellants effectively. This is limited, however, by the machining of the injector manifold and the injectors themselves. One possibility is a set of 19 injector pairs arranged as shown in figure C.2. The diameter of an individual injector is

$$D_i = \sqrt{\frac{A_i}{n\pi}}$$

so, for 19 injectors, the diameters for the fuel and oxidizer injectors, respectively, are

$$D_{i,f} = 0.979 \text{ mm}$$

$$D_{i,o} = 1.104 \text{ mm}$$

The injectors should be positioned at angles such that the net momentum of the fuel and oxidizer streams is parallel to the axis of the engine, so

$$\begin{aligned} m_o v_o \sin(\phi_o) &= m_f v_f \sin(\phi_f) \\ \phi_o &= \sin^{-1} \left[ \frac{m_f^2 D_o A_{i,o}}{m_o^2 P_f A_{i,f}} \sin \phi_f \right] \end{aligned}$$

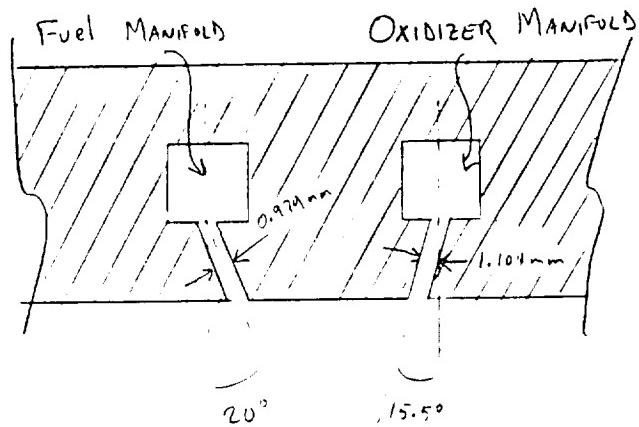
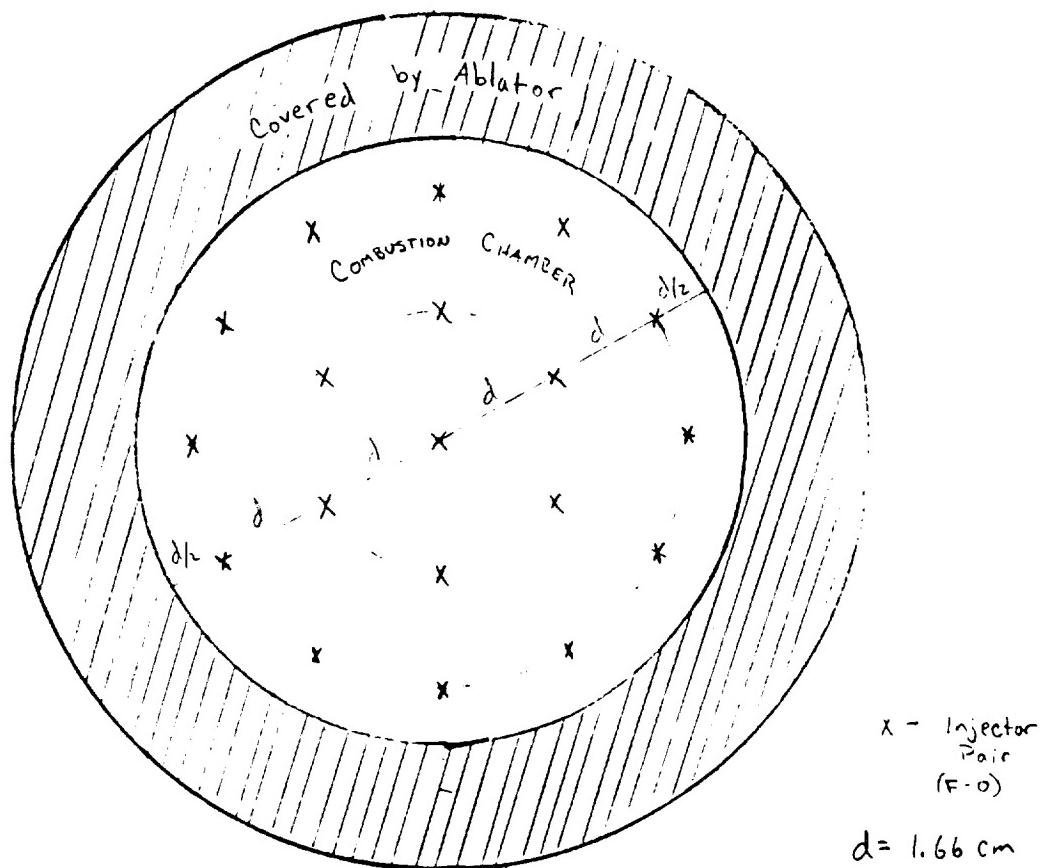


FIGURE C-2: LIQUID ENGINE INJECTOR

Open  
Flow  
Schematic

Assuming a fuel injector angle of  $20^\circ$  from the engine axis, the oxidizer injector angle should be

$$\phi_o = \sin^{-1} \left[ \left( \frac{0.203}{0.331} \right)^2 \left( \frac{1450}{910} \right) \left( \frac{1.82}{1.43} \right) \sin 20^\circ \right]$$

$$\phi_o = 15.5^\circ$$

**Chamber Stress Analysis:** The stress analysis of the combustion chamber is very straightforward, considering only the hoop stresses. For 3-Directional Carbon-Silicon Carbide Novoltex<sup>(R)</sup>,  $\rho = 2300 \text{ kg/m}^3$ ,  $\sigma_y = 80 \text{ MPa}$ , and for hoop stresses,

$$t_w = r_c (P_c / \sigma_y) \times \text{S.F.}$$

where S.F. is the safety factor (1.5) and  $r_c$  contains the ablative chamber liner. Substituting in the values,

$$t_w = (0.0445 + 0.015)(10^6 / (80 \times 10^6)) \times 1.5$$

$$t_w = 1.1 \text{ mm}$$

A combustion chamber with walls this thin would be extremely difficult to construct reliably, so an additional safety factor will be added to the wall thickness by increasing it to a uniform 2.0 mm.

**Mass Estimates:** Given the material properties and dimensions calculated above, it is very straightforward to compute the mass of the engine as the sum of the masses of the components. The volume of 3-D Novoltex<sup>(R)</sup> in the combustion chamber and nozzle shells is

$$V_{c/c} = (0.002 \text{ m}) (2\pi(r_c + 0.015 \text{ m}) + A_n)$$

Where  $A_n$ , the nozzle surface area, is given by<sup>[C-2]</sup>

$$A_n = 70 \times A_t$$

$$V_{c/c} = (0.002) (2\pi(0.0445+0.015) + 70(0.0009)) \text{ m}^3$$

$$V_{c/c} = 8.74 \times 10^{-4} \text{ m}^3$$

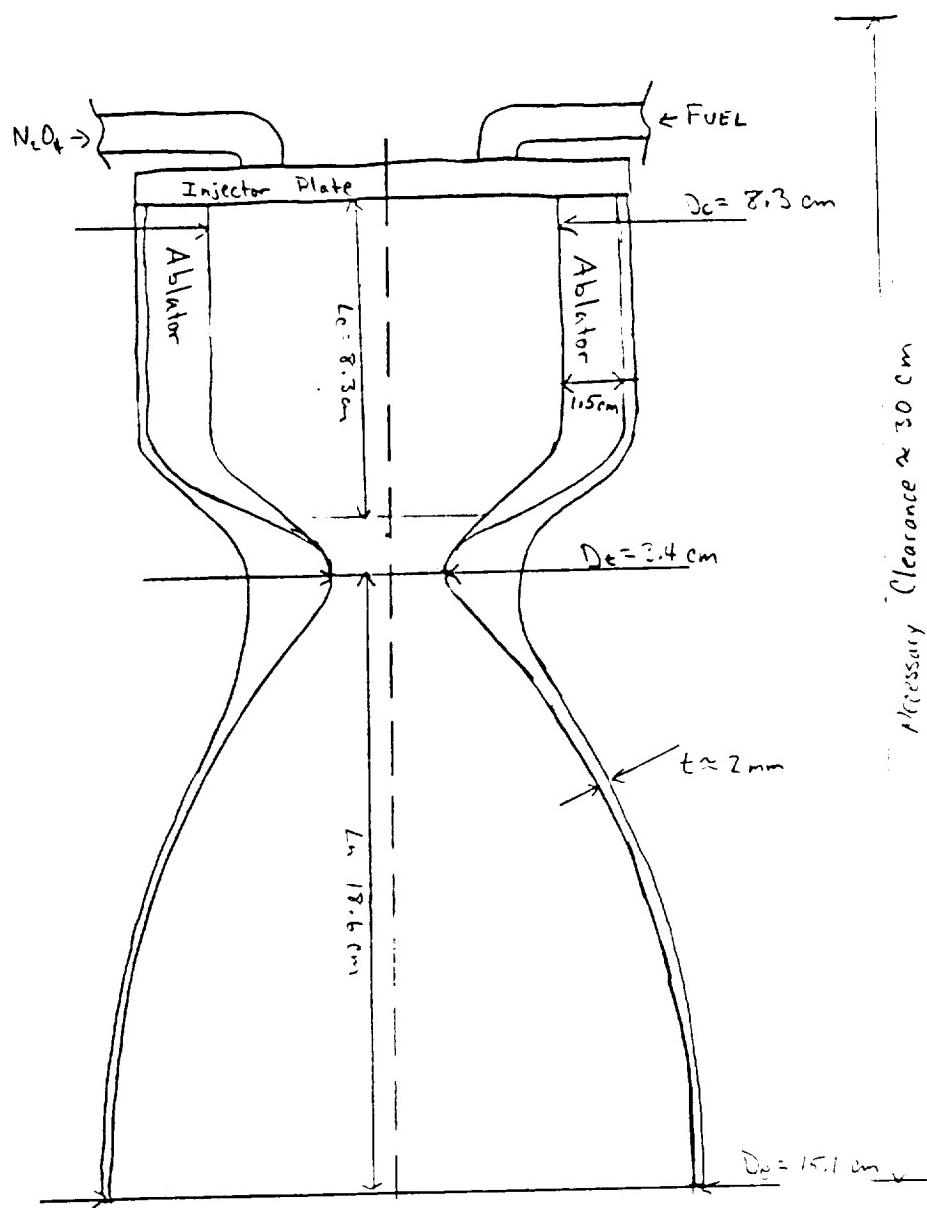


FIGURE C-3 MSP LIQUID ENGINE

So the mass of the chamber and nozzle shells is approximately

$$M_{\text{shell}} = 2300 \text{ kg/m}^3 \times 8.74 \times 10^{-4} \text{ m}^3$$

$$M_{\text{shell}} = 0.20 \text{ kg}$$

The ablative insert mass, assuming a uniform density of 1740 kg/m<sup>3</sup> is

$$M_{\text{abl}} = 2\pi(r_c + 0.015/2) \times l_c \times \rho_{\text{abl}} \times 0.015$$

$$M_{\text{abl}} = 2\pi(0.0445 + 0.0075) \times 0.0833 \times 1740 \times 0.015$$

$$M_{\text{abl}} = 0.71 \text{ kg}$$

And the mass of the injector plate, assumed stainless steel ( $\rho = 7600 \text{ kg/m}^3$ ), 1 cm thick, and 50% hollow for fuel manifolds,

$$M_{\text{inj}} = \pi(r_c + 0.015)^2 \times 0.01 \times 7600 \times 50\%$$

$$M_{\text{inj}} = 0.80 \text{ kg}$$

Thus the entire engine mass is approximately

$$M_{\text{eng,liq}} = 0.20 + 0.71 + 0.80 = 1.71 \text{ kg.}$$

Allowing some excess for throat reinforcement, plumbing, and the gimballing bearing,

$$M_{\text{eng,liq}} = 2.0 \text{ kg}$$

---

#### Solid Orbital Insertion Motor:

The analysis of the solid motor is very similar to that for the liquid motor. The properties of the solid Polyurethane/Ammonium Perchlorate/Aluminum fuel, again from Sutton and Ross<sup>[C-2]</sup>, are:

Flame Temperature:	3088°K
R of Exhaust Products:	284 J/(kg-K)
Y of Exhaust Products:	1.17
Burning Rate:	6 mm/s
Density:	1750 kg/m <sup>3</sup>

Dimensions: Assuming a chamber pressure of 1 MPa and a

nozzle exit area ratio of 20:1, the Mach number at the exit plane is found (again by computer iteration) to be 3.65. At the exit,

$$P_e = (10^6) [1 + \frac{1.17-1}{2} 3.65^2]$$

$$P_e = 5452 \text{ Pa}$$

And the thrust coefficient is

$$C_{f, \text{ideal}} = \sqrt{\frac{2(1.17)^2}{1.17-1} \left(\frac{2}{1.17+1}\right)^{\frac{1.17+1}{1.17-1}} \left[1 - \left(\frac{5452}{1000000}\right)^{\frac{1.17-1}{1.17}}\right]} + \frac{5452-0}{1000000} \quad (20)$$

$$C_{f, \text{ideal}} = 1.849$$

Again applying the 98% nozzle efficiency (similar geometry to liquid engine):

$$C_f = 1.812$$

Assuming a maximum thrust of 200 N (2.0 kg final mass at 100 m/s<sup>2</sup>),

$$A_t = \frac{F_t}{C_f P_e} = \frac{200 \text{ N}}{1.812 (10^6 \text{ Pa})}$$

$$A_t = 1.103 \times 10^{-4} \text{ m}^2$$

$$A_t = 1 \text{ cm}^2$$

So the actual maximum thrust is

$$F_{t,\text{max}} = (10^{-4})(10^6)(1.812) = 181.2 \text{ N}$$

The mass flow is again given by

$$\dot{m} = a^* A^* \rho^* = P_c A_t \sqrt{\frac{Y}{R T_c}} \left(\frac{2}{Y+1}\right)^{\frac{Y+1}{2(Y-1)}}$$

$$\dot{m} = (10^6)(10^{-4}) \sqrt{\frac{1.17}{284.368}} \left(\frac{2}{1.17+1}\right)^{\frac{1.17+1}{2(1.17-1)}}$$

$$\dot{m} = 0.0687 \text{ kg/s}$$

And hence the specific impulse:

$$I_{sp} = 181.2 / ((0.0687)(9.81)) = 269 \text{ sec}$$

The specific impulse, from the ideal rocket equation, gives the propellant mass and volume:

$$M_p = 2.0 \text{ kg} \times [\exp\left(\frac{\Delta V}{g_{\text{vac}} \rho_p}\right) - 1]$$

$$M_p = 2[\exp\left(\frac{828}{1.81/(28)}\right) - 1]$$

$$M_p = 0.734 \text{ kg}$$

$$V_p = M_p / \rho_p = 4.19 \times 10^{-4}$$

The burning area ( $A_b$ ) for these conditions can be found from

$$A_{b,\max} = \dot{m} / (\rho_p r_b)$$

$$A_{b,\max} = (0.0687) / ((1750)(0.0006))$$

$$A_{b,\max} = 65.4 \text{ cm}^2$$

**Grain Design:** For a neutral-burning, cylinder-based grain configuration (such as the conventional star grain), the cylinder length and radius, in order to meet the criteria on area and volume, must be

$$r = 2V_p/A_b = 2(4.19 \times 10^{-4}) / 65.4 \times 10^{-3}$$

$$r = 17.1 \text{ cm}$$

$$l = V_p / (\pi r^2) = 6 \text{ mm}$$

Obviously this grain configuration is absurd, so another must be sought. Since this motor operates in free fall, a neutral burn is not required, and the options for grain configurations is almost endless. This motor will have a "sphere within a sphere" grain configuration as shown in figure C.4. The burning area as a function of time is

$$A_b = \frac{\pi(r_0 + r_b t)}{r_c} \left[ r_c^2 - (r_0 + r_b t - r_c)^2 \right]$$

and the remaining propellant mass is

$$M_p = \frac{\pi \rho_p}{3} \left[ 2r_c^3 + \frac{3}{2}r_c(r_0 + r_b t)^2 - \frac{1}{8}(r_0 + r_b t)^3 \right]$$

Using the fact that the thrust in vacuum is directly proportional

to  $A_b$  and  $P_c$ , a computer spreadsheet was programmed to calculate the chamber pressure and acceleration as functions of time (figures x.x and x.x), and various dimensions were tested. The final propellant grain dimensions are a total grain radius of 4.65 cm and an initial bore radius of 1.0 cm. As can be seen from the graphs, the maximum acceleration is  $90.0 \text{ m/s}^2$  and the maximum chamber pressure is 1.25 MPa. This pressure is slightly higher than initially assumed, but easily feasible.

**Stress Analysis and Mass Estimate:** The stress analysis is nearly a duplicate of that used for the liquid engine except for the fact that this case is spherical rather than cylindrical. Chamber wall thickness is given by

$$t_w = r_c/2 \times (P_c/\sigma_y) \times S.F.$$

With similar materials,

$$t_w = (0.0465/2)(1.25/80)(1.5)$$

$$t_w = 0.545 \text{ mm}$$

This chamber is also impossible to reliably construct, so again, a 2 mm uniform wall thickness is assumed. This gives the total mass of the case and nozzle:

$$M_{tot} = (0.002)(4\pi(0.0465)^2 + 70(0.0001))(2300)$$

$$M_{tot} = 0.157 \text{ kg}$$

Allowing 0.093 kg for throat inserts, internal insulation, and attachment hardware, the mass of the solid motor is

$$M_{fueled} = 0.984 \text{ kg}$$

$$M_{dry} = 0.250 \text{ kg}$$

**Solid Motor Spin-up Mechanism (Clockspring):** In order to have stability in a spin-stabilized spacecraft, the rotational

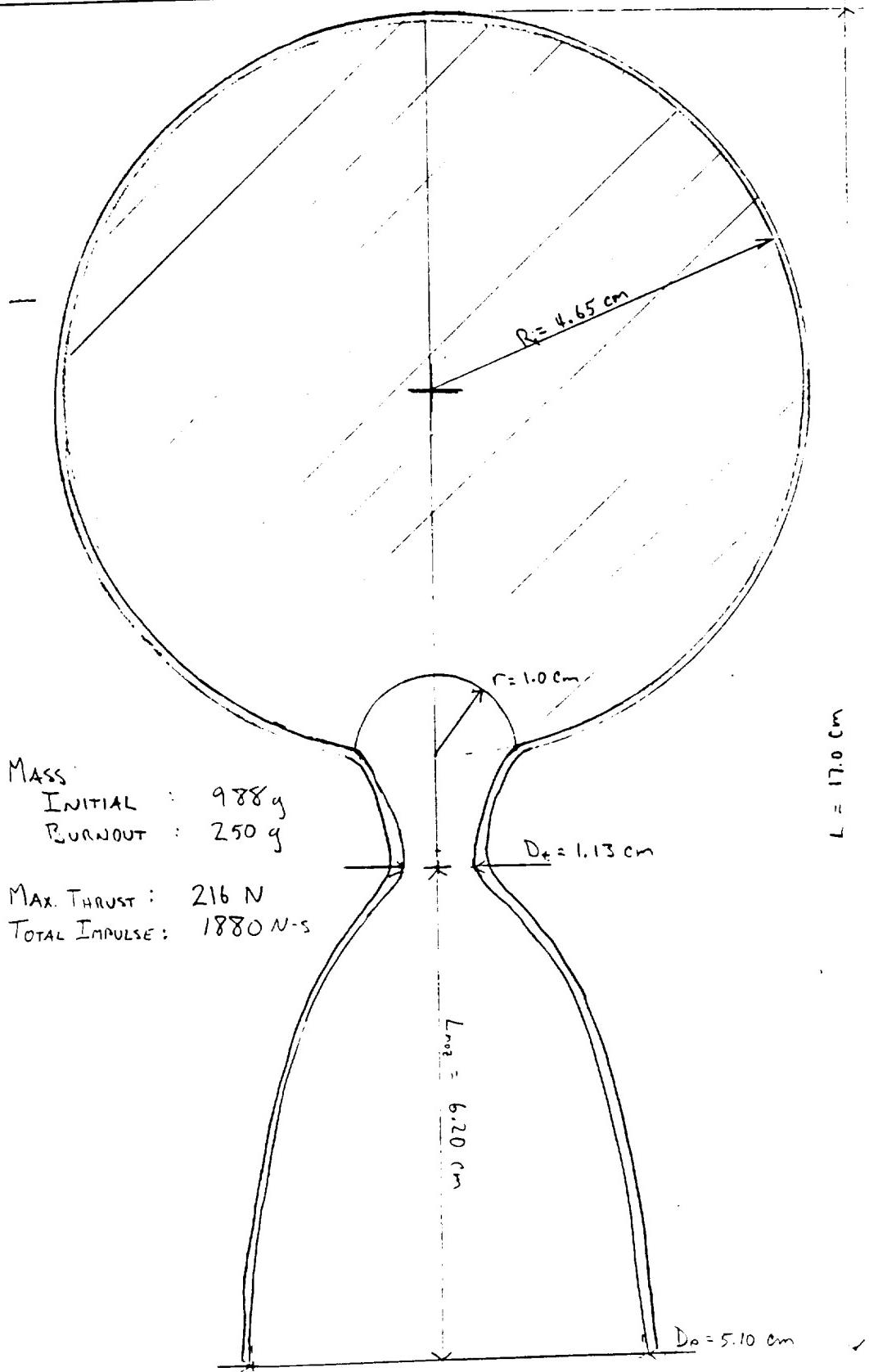


FIGURE C.4  
SOLID INSERTION MOTOR

ORIGINAL PAGE IS  
OF POOR QUALITY

FIGURE C.5: Final Stage Acceleration Profile

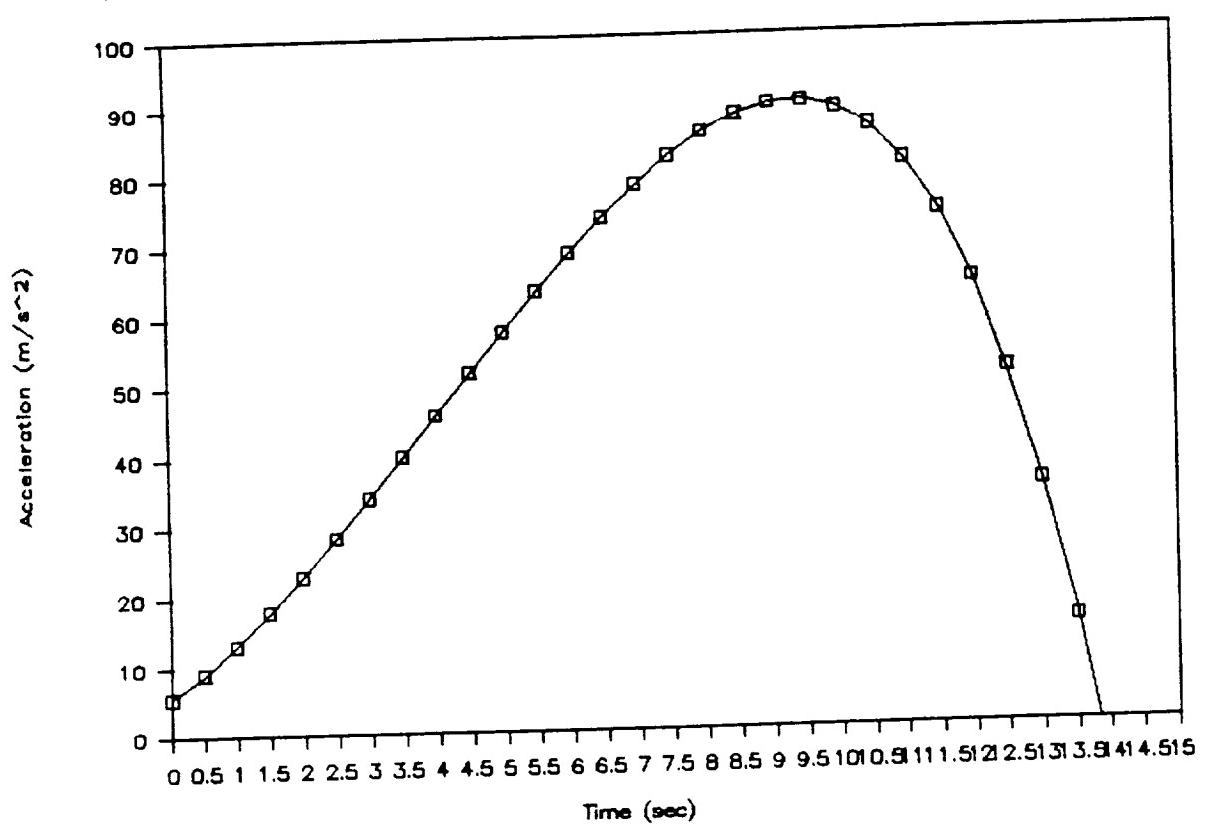
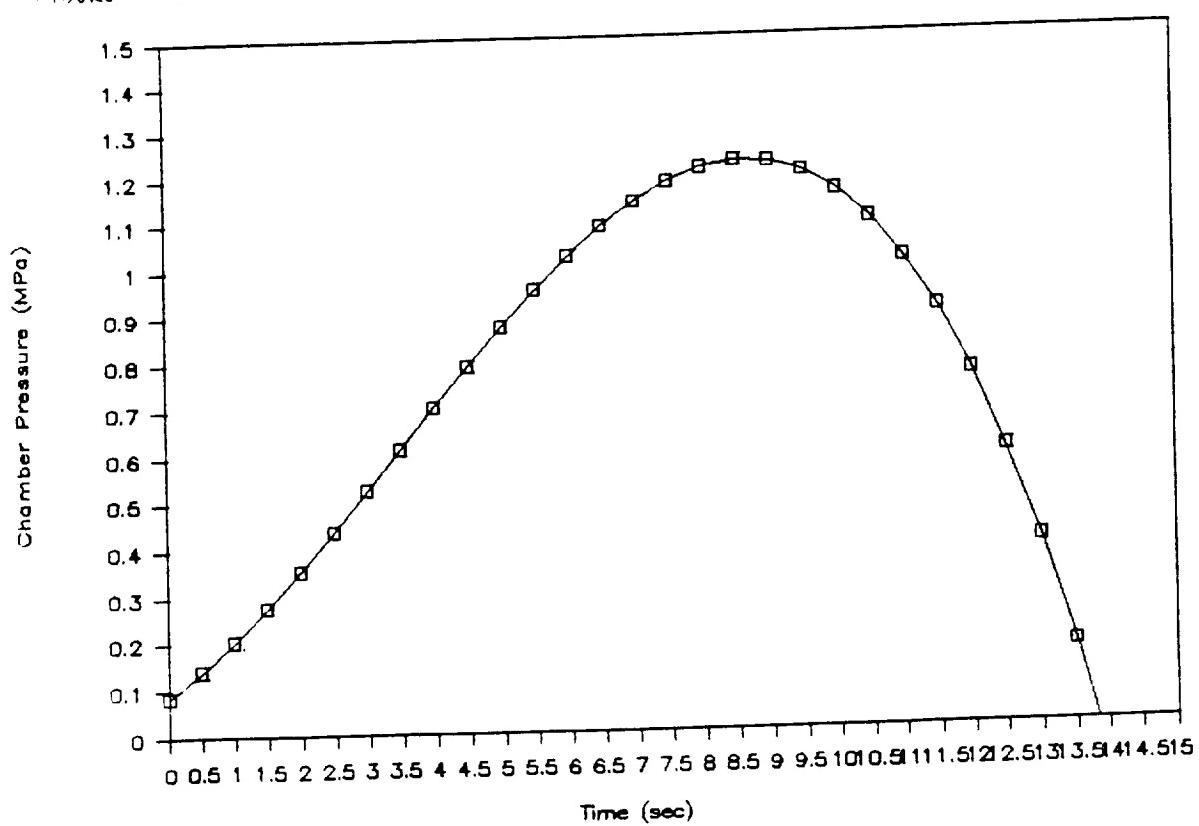


FIGURE C.6: Solid Motor Chamber Pressure Profile



kinetic energy must be much greater than the maximum anticipated disturbing moment. Assuming a 1 mm misalignment of the thrust vector with the center of mass at 200 N thrust, the disturbance torque is

$$M_d = (0.001 \text{ m})(200 \text{ N}) = 0.2 \text{ N}\cdot\text{m}$$

The rotational kinetic energy of a spinning body is given by

$$E_k = I \omega^2 / 2$$

where  $I$  is the moment of inertia about the spin axis and  $\omega$  is the spin rate in radians per second. Assuming the rendezvous stage to be a uniform cylinder of  $r = 5 \text{ cm}$ ,  $m = 2.7 \text{ kg}$ , then  $I$  is given by

$$I = mr^2/2 = (2.7 \text{ kg})(0.05 \text{ m})^2/2$$

$$I = 0.00375 \text{ kg}\cdot\text{m}^2$$

Assuming a kinetic energy of five times the disturbance torque,

$$\omega = \sqrt{\frac{10(0.2 \text{ N}\cdot\text{m})}{0.00375 \text{ kg}\cdot\text{m}^2}}$$

$$\omega = 23.09 \text{ radians/sec}$$

with a safety factor of over 2 to include the depletion of propellant mass,

$$\omega = 50 \text{ radians/sec}$$

The angular acceleration of a rotational mass is

$$\alpha = \frac{T}{I} = \frac{d^2\theta}{dt^2}$$

and the torque exerted by a rotational spring is

$$T = -k\theta$$

which implies

$$\frac{d^2\theta}{dt^2} + \frac{k}{I}\theta = 0$$

$$\theta = -\theta_0 \cos(\sqrt{\frac{k}{I}} t)$$

$$\omega = \frac{d\theta}{dt} = \theta_0 \sqrt{\frac{k}{I}} \sin(\sqrt{\frac{k}{I}} t)$$

At the maximum rotational velocity, this is

$$\omega_{\max} = \theta_0 \sqrt{\frac{K}{I}}$$

$$k = \left(\frac{\omega_{\max}}{\theta_0}\right)^2 I$$

Assuming a spin-up angle of 10 revolutions (62.8 radians),

$$k = \left(\frac{50}{62.8}\right)^2 (0.00375)$$

$$k = 0.00238 \text{ N-m/rad}$$

This is a very small torque, and a spring to provide it, along with a bearing and shaft, should weigh no more than 0.1 kg.

---

#### Liquid Propellant Tanks:

**Descent:** During the descent phase of the mission, there are three separate liquid motor firings: Orbital Maneuvering, De-orbit, and Retro Landing. The mass of fuel required for each burn is calculated by applying the ideal rocket equation,

$$M_{fuel} = M_{veh} [\exp\left(\frac{\Delta V}{g_0 \bar{I}_{sp}}\right) - 1]$$

using the V's and  $M_{veh}$ 's at the given time. Applying the equation,

$$M_{f,om} = 0.802 \text{ kg}$$

$$M_{f,d-o} = 7.353 \text{ kg}$$

$$M_{f,ret} = 1.850 \text{ kg (max)}$$

$$M_{f,desc} = 10.105 \text{ kg}$$

Allowing almost 2 kg of fuel for ullage and contingencies such as an extended retro firing, the total descent fuel mass is 12 kg.

Using the mixture ratio and densities of the propellants, the necessary tankage is 5.1 liters each of  $N_2O_4$  and 50-UDMH. The volume of Helium necessary to keep these tanks pressurized to 1.2 MPa can be calculated by modelling the process as an adiabatic

expansion of an ideal gas where

$$PV^\gamma = \text{Constant}$$

For an adiabatic expansion from  $P_1$ ,  $V_1$  to  $P_2$ ,  $V_2$ , the required  $V_1$  is

$$V_1 = V_2 \times P_1^{1/\gamma} / (P_2^{1/\gamma} - P_1^{1/\gamma})$$

$$V_{He} = (2 \times 5.1) \times 1.2^{1/1.66} / (25^{1/1.66} - 1.2^{1/1.66})$$

$$V_{He} = 2.0 \text{ liters}$$

Taking three identical sets of tanks, the radii are

$$r_f = 7.4 \text{ cm}$$

$$r_o = 7.4 \text{ cm}$$

$$r_{He} = 5.4 \text{ cm}$$

Using 2014-T6 Aluminum ( $\rho = 2800 \text{ kg/m}^3$ ,  $\sigma_y = 386 \text{ MPa}$ ) for the propellant tanks and Ti-6Al-4V Titanium ( $\rho = 4430 \text{ kg/m}^3$ ,  $\sigma_y = 999 \text{ MPa}$ ) for the Helium, the required wall thicknesses are, from an analysis identical to that used in the spherical solid rocket case:

$$t_f = 0.17 \text{ mm}$$

$$t_o = 0.17 \text{ mm}$$

$$t_{He} = 1.01 \text{ mm}$$

The thicknesses of the fuel and oxidizer tanks are thinner than can be practically manufactured, so assuming a minimum machineability of aluminum to be 0.3 mm, the total tank mass is

$$M_t = 12\pi(r_f^2 t_f \rho_{Al} + r_o^2 t_o \rho_{Al} + r_{He}^2 t_{He} \rho_{Ti})$$

$$M_t = 12\pi((2)(2800)(0.074^2)(0.0003) + (4430)(0.054)^2(0.00101))$$

$$M_t = 0.84 \text{ kg}$$

Adding over 50% for valves and plumbing, the total fuel system mass for the descent stage is  $M = 1.50 \text{ kg}$ .

**Ascent:** The fuel requirements for the ascent burn cannot be calculated from the ideal rocket equation due to the effects of aerodynamic drag, but since the burn time and thrust are known, the mass of fuel expended can be calculated using the specific impulse:

$$M_p = \frac{F_t t}{g_0 I_s p}$$

Where  $t_b$  is the burn time in seconds. For the ascent stage, with a 1500 Newton constant thrust for 82 seconds,

$$M_p = (1500)(82)/((9.81)(298))$$

$$M_p = 42.0 \text{ kg}$$

Adding 5% for ullage,

$$M_p = 44.1 \text{ kg}$$

Using a breakdown similar to the descent tanks, this requires volumes of

$$V_f = 18.5 \text{ liters}$$

$$V_o = 18.9 \text{ liters}$$

$$V_{He} = 7.3 \text{ liters}$$

In order to minimize the envelope of the ascent vehicle, the tank configuration shown in figure C.7 was selected. The spherical caps and the cylindrical section have a radius of

$$R = \sqrt[3]{\frac{3V_f}{4\pi}}$$

$$R = 18.5 \text{ cm}$$

and the cylindrical section (containing  $N_2O_4$ ) has a length of

$$L = V_o / (\pi R^2)$$

$$L = 18.0 \text{ cm}$$

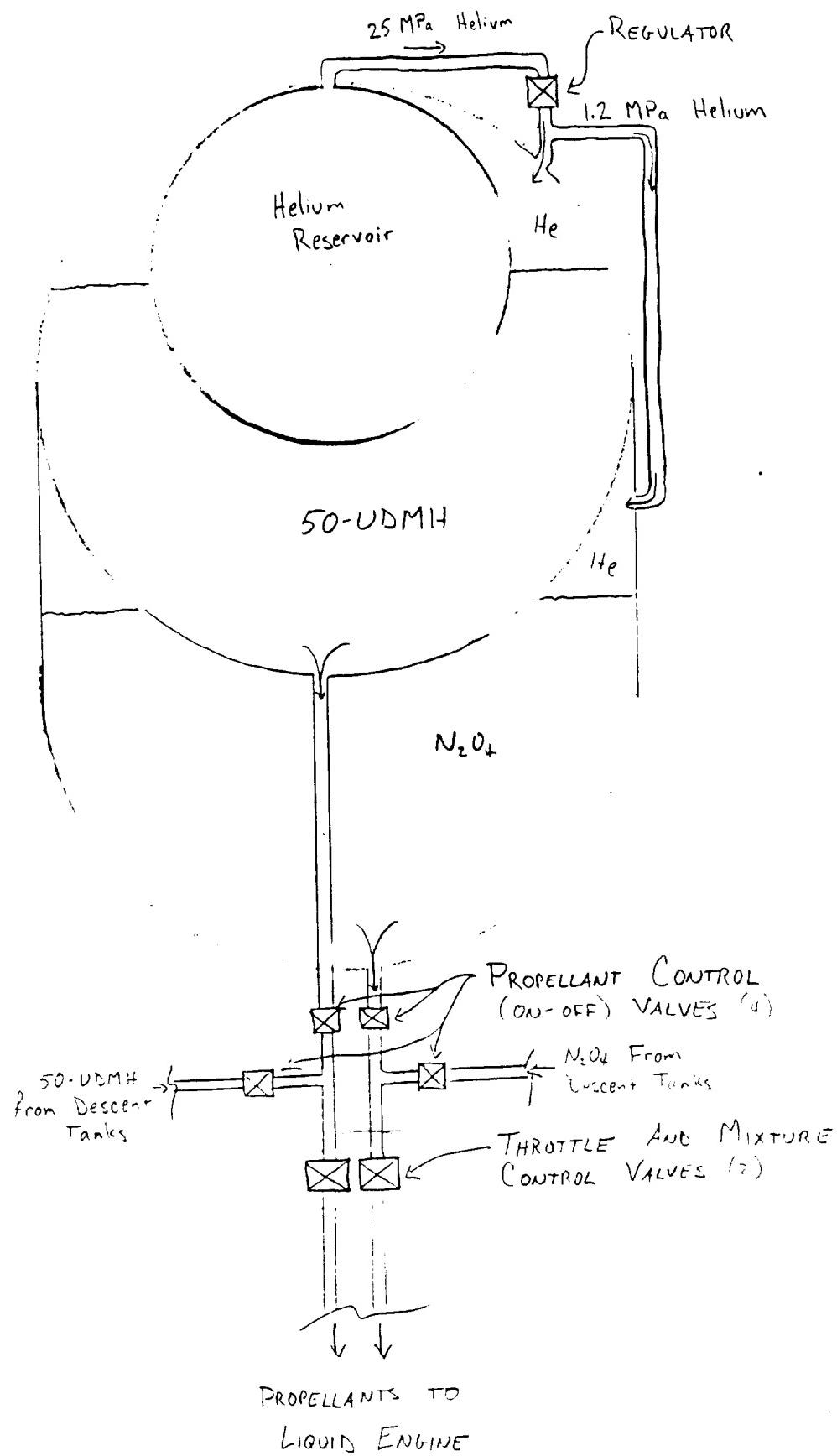


FIGURE 1-7  
ASPIENT PROPELLANT TANKS AND PUMPING

ORIGINAL PAGE IS  
OF POOR QUALITY

The Helium tank has radius

$$R_{He} = \sqrt{\frac{3(0.0073)}{4\pi}}$$

$$R_{He} = 12.0 \text{ cm}$$

Again using 2014-T6 Aluminum for propellant tanks and Ti-6Al-4V Titanium for the Helium, the required thicknesses are:

$$t_{sphere} = 0.43 \text{ mm}$$

$$t_{cyl} = 0.86 \text{ mm}$$

$$t_{He} = 2.26 \text{ mm}$$

The total mass of the ascent tank structure is

$$M = \pi((6R^2t_{sphere} + 2RLt_{cyl})(\rho_{Al}) + 4R_{He}^2t_{He}\rho_{Ti})$$

$$M = \pi(6(0.185)^2(0.00043) + 2(0.185)(0.180)(0.00086)) \\ \times (2800) + \pi(4)(.120)^2(0.00226)(4430)$$

$$M = 3.07 \text{ kg}$$

Adding 0.43 kg for plumbing and valves, the final tank mass is

$$M = 3.50 \text{ kg}$$

References:

---

- C-1: Paul Donguy and Jacques Broca, "High Temperature Composite Materials for Rocket Propulsion," AGARD Conference Proceedings #449, 1988.
- C-2: George P. Sutton and Donald M. Ross, Rocket Propulsion Elements: An Introduction to the Engineering of Rockets, John Wiley and Sons, N.Y., 1976.
- C-3: J. L. Tuttle and D. H. Blount, Perfect Bell Nozzle Parametric and Optimization Curves, NASA Reference Publication 1104, 1983.
- C-4: Liquid Rocket Engine Self-Cooled Combustion Chambers, NASA Technical Report N78-21211, 1978.
- C-5: Elliot Ring, Rocket Propellant and Pressurization Systems, Prentice-Hall, N.J., 1964
- C-6: F. A. Williams, M. Barrere, and N. C. Huang, Fundamental Aspects of Solid Propellant Rockets, AGARD, 1969.
- C-7: Pressurization Systems for Liquid Rockets, NASA Technical Report N76-22300, 1976.

Section D:  
**Orbital Mechanics**

Bob Langberg

ORBITAL TRANSFER AND ASCENT/DESCENT TRAJECTORIES

PLANETARY CHARACTERISTICS PERTINENT TO ORBITAL CALCULATIONS

Mass	6.4182 $10^{23}$ (kg)
Universal Gravitational Constant	6.6732 $10^{-11}$ ( $m^3/kg\ s^2$ )
$\mu$ Gravitational Constant for Mars	4.2830 $10^{13}$ ( $m^3/s^2$ ) 42830. ( $km^3/s^2$ )
Equatorial Gravitational Acceleration	3.73 ( $m/s^2$ )
Mean Radius	3402 ( $km$ )
Sidereal Day	24 hours 37 minutes 23 seconds 24.623 (hours) 1477.4 (minutes) 88643. (seconds)
Angular Velocity	0.24367 (deg/min) 4.2529 $10^{-3}$ (rad/min)
Rotational Speed (Equatorial)	14.468 ( $m/min$ ) 0.24113 ( $m/s$ )
(28.5° Latitude)	12.715 ( $m/min$ ) 0.21191 ( $m/s$ )

MARS ORBITING VEHICLE (MOV) ORBITAL PARAMETERS

Circular  
Ascending Node at 250° Longitude  
Inclination 30°  
Altitude 600 ( $km$ ) above mean radius  
Orbital Radius 4002 ( $km$ )  
Orbital Velocity 3.2715 ( $km/s$ )  
Orbital Period 2 hours 8 minutes 6 seconds  
2.1351 (hours)  
128.11 (minutes)  
7686.4 (seconds)

LANDING SITE

Depression Hellas, near Crater 29  
Altitude 4.2 ( $km$ ) Below Mean Radius  
Radius of Site 3397.8 ( $km$ )  
Longitude 283 (Deg)  
Latitude -28.5 (Deg)

### DESCENT ANALYSIS

#### IDEAL DESCENT ANALYSIS, NEGLECTING ATMOSPHERIC DRAG

Initially the descent trajectory was modeled without taking the atmosphere into account, using ideal equations for elliptical orbital motion (Kepler Motion). In this analysis there was an orbital plane change, calculated with spherical trigonometry, at the same time as the deorbit burn. This combined burn put the Mars Surface Probe (MSP) on an elliptical orbit that intersects the Martian Surface at the landing site. It was optimized with the constraint that the flight angle at landing could not be less (not shallower) than  $15^\circ$ . Orbits of eccentricity varying from 0.08 (nearly circular) to 0.93 (an extremely narrow ellipse) were examined. The delta v required was least at the more circular ellipses, with an absolute minimum occurring at the ellipse with eccentricity of 0.14, which had the minimum plane change angle ( $10.1^\circ$ ), unfortunately its angle with the horizon upon landing was  $7.83^\circ$  which was too shallow.

The best case which did meet the  $15^\circ$  requirement was the transfer ellipse with eccentricity of 0.270 . It had a velocity at apoapse (4002 km) of 2.795 km/sec, which is 0.476 km/sec less than the MOV's circular orbital velocity of 3.271 km/sec. This orbit takes the MSP through a  $58.7^\circ$  change in true anomaly, to land with an impact velocity of 3.409 km/sec. The entire deorbit takes 21.8 minutes, during which time Mars rotates  $5.31^\circ$  on its axis. The plane change angle for this case was  $11.7^\circ$ , which when combined with its velocity change for deorbit (0.476 km/sec) required a total delta v of 0.781 km/sec. This was the best that could be done with the plane change while neglecting the atmosphere.

## DETAILED DESCENT ANALYSIS, WITH ATMOSPHERIC DRAG

### *Avoiding the Plane Change Maneuver*

After the ideal analysis was completed it was decided that a significant fuel savings (305 m/sec) could be attained by avoiding the 11.7° plane change. By leaving the MOV orbit with a small delta v of 8.1 m/s (0.2% of the circular velocity) the MSP would advance or recede by a rate of 2.5° per orbit. After 72 orbits, the MSP may move up to ±180°, allowing it to be positioned anywhere on the orbit. With precise positioning it may then deorbit as Mars rotates and the target point passes below, without the necessity of a plane change. This positioning would take up to 6.2432 Mars sidereal days (6 sidereal days and 9 hours) and would allow the MSP to position itself at any point in orbit to successfully deorbit and land at the specified sampling site. The extended time in orbit presents no apparent thermal problems.

### *Limits of Atmospheric Effects*

To accurately model the effects of the atmosphere, the deorbit trajectory was assumed to be a Kepler ellipse until an altitude of 250 km was attained. The drag force on the MSP at an altitude of 250 km is  $1.04 \cdot 10^{-4}$  N. The maximum drag is  $6.92 \cdot 10^{+3}$  N which occurs at an altitude of 30.1 km. This difference, in excess of seven orders of magnitude, shows that not only is it a very reasonable assumption to ignore atmospheric effects above 250 km, but that atmospheric drag probably could have been ignored until about 150 km altitude where the drag is approximately 0.6 N.

### *Incremental Time Step Analysis*

At 250 km, the velocity and flight path angle was computed for the ideal transfer orbit. These were used as initial conditions for a time stepping analysis. In this analysis, the aerodynamic force on the MSP, which was assumed to be drag only, with no lift is computed. The drag

force was computed based on a Cd of 1.0 which is referenced to the projected area of the MSP. The gravitational acceleration was computed as a position of radius using the standard inverse square law at each position during the descent. These two forces were used to compute the acceleration in the radial and tangential directions and the resultant changes in velocity and position. Time increments ranged from one second to ten seconds, with five second increments used during the period of maximum aerodynamic forces. The MSP mass was changed as the heat shield is dropped as the parachute is deployed. The Cd reference area was changed at the time of parachute deployment, but the Cd is kept at 1.0. Both of these changes take place approximately 1.2 km above the landing site.

#### *Terminal Velocity of Heat Shield and Parachute*

Terminal velocity design charts were generated for the heat shield and parachute design analysis. These charts were used in the sizing process only and were not directly part of the incremental time/dynamic analysis. The final dimensions for the heat shield and parachute diameters are 1.5 and 4.0 m respectively.

#### *Atmospheric Model*

The atmospheric model used to calculate the variation of density with altitude, used for drag calculations is based on: *The Mars Reference Atmosphere, 1982*, published by The Committee on Space Research, Chapter 1: "Post-Viking Models for the Structure of the Summer Atmosphere of Mars", written by A. Seiff. Data from this source concerning the atmosphere in the southern hemisphere, during summer, at low altitudes was used to construct an exponentially decaying atmosphere model. The model is based on the density at the mean surface,  $1.78 \times 10^{-2} \text{ kg/m}^3$ , and an exponential scale height, 11.75 km, where the density has fallen off by  $1/e$ .

$$\text{density(alt)} = \text{density}(0) * \exp[-\text{alt}/\text{scale ht.}]$$

This model is most accurate where it has the greatest influence on the MSP, at low altitudes where the density is greatest, and the resulting aerodynamic forces are largest. See graphs for the shape of the atmospheric models. It is significant to note that the density has fallen off the linear scale at 70 km altitude. (0.26% of the density at the surface.

*Optimizing Deorbit Delta V with respect to Eccentricity*

The transfer ellipse that sets the initial conditions for the descent analysis was originally assumed to be the transfer ellipse from the ideal case with an eccentricity 0.27. Due to the drag on the heat shield and parachute, the landing trajectory is nearly vertical near the surface. Thus the 15° restriction on landing angle is no longer a constraint for the real descent case.

By doing a computation comparing the deorbit fuel required vs. the eccentricity of the exoatmospheric transfer ellipse, a simple relationship was developed. It shows that the delta v only depends on the original circular velocity and the transfer ellipse eccentricity, e:

$$\Delta V = V_{\text{Circular}} * [ 1 - \sqrt{1-e} ]$$

This function is tabulated and graphed. It was found that another significant fuel savings could be realized by making an even shallower approach to the Martian surface, and the shallower the approach, the more fuel saved.

The problem with making the approach more shallow is that the distance traversed becomes greater, and the accuracy of the reentry is reduced. For this reason the transfer ellipse was made only slightly more circular. The original eccentricity was 0.27, requiring a delta v of 0.476 km/sec, and the new eccentricity is 0.10, requiring a delta v of only 0.169 km/sec. This causes the true anomaly traversed in descent to increase from 53.42° to 98.10°, and

the time of descent to increase from 22.37 to 36.04 minutes. However, most of this increased travel is exoatmospheric. The atmospheric true anomaly traversed (250 km to surface) increases from  $11.23^\circ$  to  $16.00^\circ$ , and the time in the atmosphere (250 km to surface) goes from 5.85 to 7.19 minutes.

These two trajectories are essentially the same near the surface where the velocities and flight path angles are: 161.7 vs. 159.9 m/sec and  $66.8^\circ$  vs.  $68.8^\circ$ , for the old and new cases respectively. These values are for an altitude 5 km above the surface. Thus, for this analysis they are essentially compatible, and since the  $e = 0.10$  case is more fuel efficient, (0.169 vs. 0.476 km/sec, a 64% savings) it was chosen as the final descent trajectory. The fuel savings was approximately 10 kg.

#### *Range Error Due to Uncertainty in Deorbit Delta V*

Because we enter the atmosphere at a rather shallow angle, ( $17.6^\circ$  at 100 km altitude), the accuracy of the trajectory was questionable. Since the vehicle is in free flight with only small attitude control thrusters for orbital corrections, it is of importance to know how sensitive the trajectory is to the only parameter we can effectively control, the deorbit delta v. The same time stepping analysis was run for a 2% decrease in delta v. The new case placed the MSP 13.64 km short of its original target, but this is a small change considering the fact that the deorbit trajectory traverses over 5800 km.

The final descent trajectory has the following characteristics:

Eccentricity 0.10  
True Anomaly Traversed (exoatmospheric/atmospheric)  
82.1°/16.0°

Radius at Apoapse 4002 km  
V Apoapse 3.102 km/sec  
V Terminal (300 m above surface) < 45 m/sec  
Delta V (deorbit) 168 m/sec  
Delta V (combined deorbit & retrofire) 211 m/sec

Max. Mach (100 km alt.) 19.3  
Max. Dyn. Press. (30.1 km alt.) 1968. N/m<sup>2</sup>  
Max. Deceleration (30.1 km) 8.2 earth g's  
Time during max deceleration (3000/250 m/s) 75 sec

### ASCENT ANALYSIS

The ascent was modeled the same way as the descent. The assumed Cd was again 1.0 . The gravitational acceleration and the atmospheric density varied with radial position. Because the 82 second 1500 N ascent burn cannot be assumed to be impulsive for a ten minute flight, the thrust was considered to act in a constant manner and was just considered to be another force on the body for the duration of the burn. It was assumed that the mass decreased linearly during this period. The resulting acceleration was computed each second during the burn and every five or ten seconds after burn out. These accelerations were used to calculate the new velocity and position in polar coordinates, just as they were in the descent analysis. The basic parameters that were varied were the length of the 1500 N burn and the launch angle. From these initial conditions, the delta v required at 600 km to circularize the orbit at the same velocity as the MOV was computed.

#### *Optimum Ascent Angle and Minimum Insertion Delta V*

The required delta v was quite sensitive to launch angle, as measured from the horizon. The optimum angle varied between  $60^\circ$  and  $62^\circ$  in a nonlinear fashion, depending on the duration of the ascent burn. At each burn duration, the optimum angle was determined to within  $0.5^\circ$ . By launching at  $0.5^\circ$  too low an angle, the ascent stage fell approximately 25 km short of its 600 km target altitude. If launched at  $0.5^\circ$  too great an angle, it would reach 600 km with a flight path angle of about  $10^\circ$ , which must be corrected for with a larger delta v for circularization. The optimum launch angle allows the ascent stage to just reach the 600 km mark with a flight path angle as close to zero as possible to minimize the delta v required.

It was also found that with longer launch burns the velocity at 600 km was greater (at the optimum launch angle), requiring less and less insertion/circularization delta v. The penalty here is the increased launch mass for fuel and tanks. There is obviously an optimum distribution of the increased mass between the boost stage and the apoapse stage, but we had a driving design condition other than total mass to consider.

The driving criteria for the launch trajectory turned out to be the apoapse kick solid rocket motor. With its burn characteristics determined by the spherical grain configuration, the maximum delta v it could provide without exceeding 10 g's was between 800 and 850 m/s. With our original launch mass of 46.4 kg the minimum possible delta v at the optimum launch angle was 1481. m/s, which could not safely be considered. A number of alternate cases were investigated.

The final optimum launch profile increased the launch mass to 56.8 kg (a 10.4 kg increase), neglecting the increase in fuel tank mass. This profile required a launch angle of 61.0° and a burn for 82 sec of 1500 N. The ascent stage reaches 600 km altitude after 10.8 minutes, with a flight path angle of only 2.92°. The required delta v is 828.0 m/sec.

If the flight path angle had been 61.5° the required delta v would have been 930.4 m/sec (a 102.4 m/s increase), and if launched at 60.5° the ascent stage would have fallen about 40 km short of its 600 km target.

#### Rendezvous Strategy

For rendezvous purposes the ascent stage could be placed into an orbit with a slightly different period than that of the MOV, thus letting the sample return vehicle catch up to the MOV without the MOV having to burn any additional fuel (passive recovery). As discussed for the deorbit positioning, a ±8.1 m/sec delta v would let the

ascent stage advance/recede by a rate of  $2.5^\circ$ /orbit. This allows for rendezvous in less than seven days. The ascent stage is not thermally sensitive at this point in the mission, therefore the rendezvous passage presents no thermal problems.

The final ascent trajectory has the following characteristics:

Ascent Thrust (82 seconds)	1500 N
Delta V Launch	4020.3 m/sec
Maximum Acceleration at Launch	10 g's
Launch Angle	61.0
Max. Mach (75 km alt.)	16.8
Max. Dyn. Press. (10 km alt.)	2732. N/m <sup>2</sup>
True Anomaly Traversed (exoatmospheric/atmospheric)	1.64° / 2.37°
Time in Atmosphere (up to 250 km altitude)	196 sec
Flight Path Angle at Insertion	2.92°
V (600.9 km Altitude)	2456.1 m/sec
V Circular (MOV)	3271.4 m/sec
Delta V insertion	828.0 m/sec
Delta V (combined launch & insertion)	4848.3 m/sec
Maximum Acceleration at Insertion	< 10 g's
Total Delta V: (m/s)	
Positioning	8.1
Deorbit	167.9
Retrofire	42.9
Ascent	4020.3
Insertion	828.0
Rendezvous	<u>8.1</u>
	5075.3

## Orbit and Trajectory Supplemental Information

Mission Overview Diagram

Mission Profile

Ideal Descent

Equations

Plane Change Calculation

Time Elapsed During Orbital Motion Calculation

Detailed Analysis

Force Balance Diagram

Pertinent Equations

Iterative Time Stepped Procedure

Used in Spread Sheet Calculations

Selected Results From Descent Trajectory, Including:

Parachute Deployment

Retro Fire Requirements

Mars Atmosphere Model

Linear Plot

Logarithmic Plot

Terminal Velocity Design Charts

Heat Shield

Parachute

Deorbit Delta V vs. Eccentricity

Graph

Table

Final Descent Profile ( $e = 0.10$ ) Spreadsheet

Graphs

Descent Altitude vs. Time

Descent Velocity vs. Time ( $e = 0.10$ )

Descent Velocity Vs. Time ( $e = 0.27$ )

Ascent Analysis

Minimum Insertion Delta V vs. Lift-Off Mass

At Optimum Launch Angle

Graph

Chart

Final Ascent Profile Spreadsheet

Launch Delta V Due To Constant 1500 N Thrust

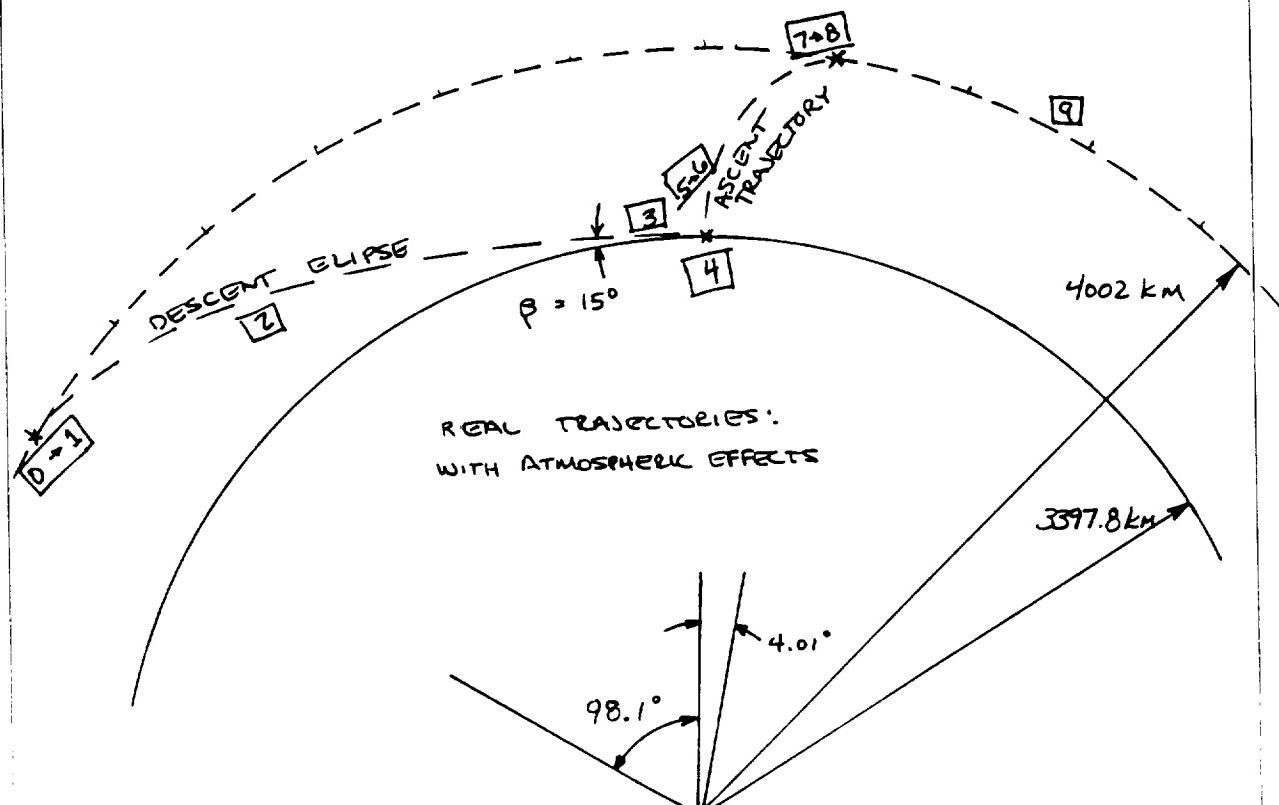
Calculation

Mach Number vs. Altitude for Ascent and Descent

Graph

Chart (Includes Dynamic Pressure)

- 0 → 1 DE-ORBIT BURN (LIQUID)  
 2 IMMEDIATELY BEFORE PARACHUTE DEPLOYMENT (VERY NEARSURFACE)  
 3 IMMEDIATELY PRECEDING IMPACT  
 4 GROUND CONDITION  
 5 → 6 ASCENT BURN (LIQUID)  
 7 → 8 APOGEE KICK BURN (SOLID) AND RENDEZVOUS  
 9 RENDEZVOUS



DISTANCES & ANGLES NOT TO SCALE.

## MISSION PROFILE

- DETACH FROM M.O.V. AND GIVE SMALL BURN FOR POSITIONING.  
~ 8.0 m/s
- SPEND UP TO SEVEN DAYS POSITIONING, M.S.P. FOR DE-ORBIT  
BURN (LIQUID ENGINES) - MODIFY POSITION UP TO 180° TO AVOID  
PLANE CHANGE
- DESCENT AND LANDING (PARACHUTE AND LIQUID ENGINE RETRO-FIRE)
  - MUST BE ON SURFACE AT LEAST ~12 HOURS UNTIL  
ORBITAL PLANE PASSES OVERHEAD.
  - COMPLETE SAMPLING MISSION DURING THIS TIME
- ASCENT WITH LIQUID ENGINE. (SAME AS USED FOR DE-ORBIT  
AND DESCENT RETRO-FIRE) LEAVE BEHIND SIGNIFICANT FRACTION  
OF VEHICLE MASS.
- APOGEE KICK WITH SOLID TO SEPARATE FROM ASCENT  
MOTORS AND TANKS, AND REACH ORBIT NEAR THE  
CIRCULAR ORBIT OF M.O.V. (AT VELOCITY SLIGHTLY  
GREATER OR SLIGHTLY BELOW THAT OF M.O.V., FOR  
SUBSEQUENT RENDEZVOUS.)

## DESCENT

## IDEAL ORBIT CALCULATIONS

1. ECCENTRICITY  $e$  TABULATED2. SEMI MAJOR AXIS  $a$   
(km)3. APOGEE VELOCITY FOR  
DEORBIT TRAJECTORY  
(km/sec)4. ANGULAR MOMENTUM OF  
DESCENT ORBIT  
(km<sup>2</sup>/sec)5. PARAMETER FOR POLARISATION.  $P$   
OF DESCENT ORBIT  
(km)6. THETA ( $\theta$ ) TRUE ANOMALY  
(degrees)7. "IMPACT" VELOCITY  
(km/sec)8. LANDING (FLIGHT PATH) ANGLE  
(degrees)

$$a = \frac{r_{\text{apogee}}}{(1+e)}$$

 $r_{\text{apogee}} = 4002 \text{ km}$ 

$$v_a = \sqrt{\mu \left[ \frac{2}{r_a} - \frac{1}{a} \right]}$$

$$r_a = r_{\text{apogee}} \quad \mu = 4.283 \cdot 10^4 \frac{\text{km}^3}{\text{sec}^2}$$

$$h = r_{\text{apogee}} v_{\text{apogee}}$$

$$P = \frac{h^2}{\mu}$$

$$\theta = \cos^{-1} \left[ \frac{P - r}{r_e} \right]$$

$$\text{From: } r = \frac{P}{1 + e \cos \theta}$$

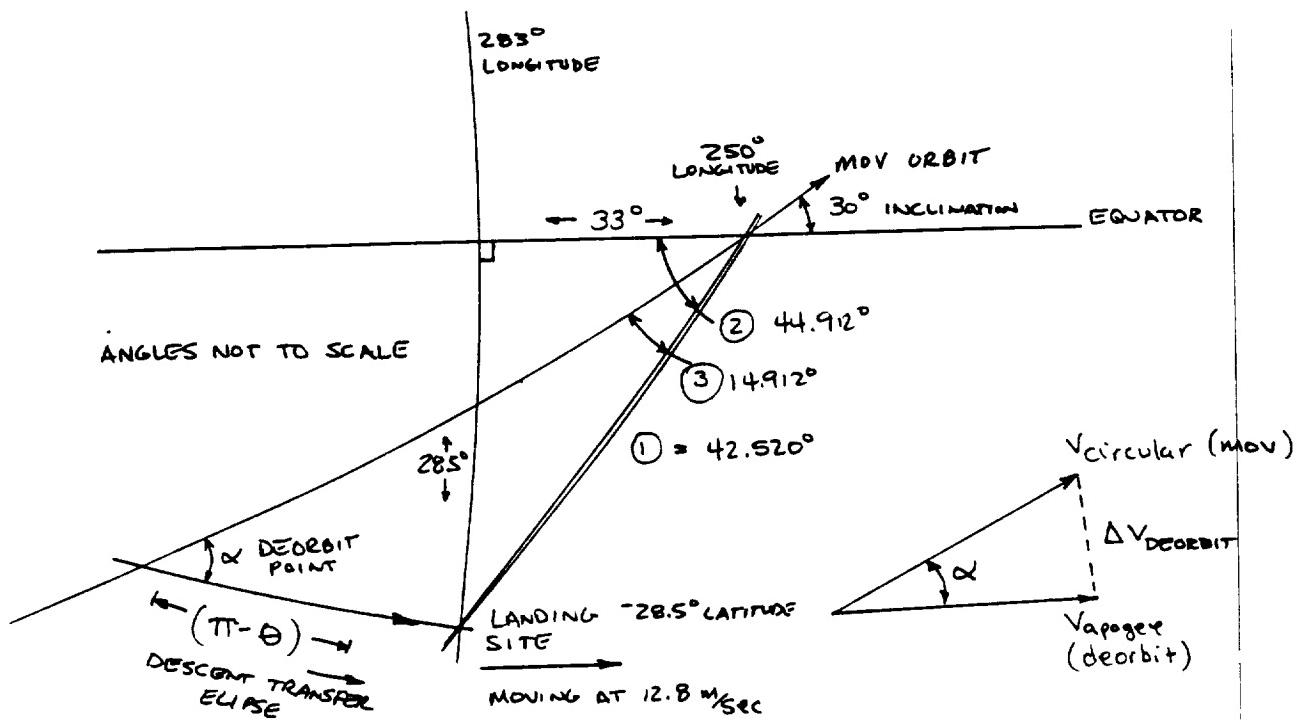
$$v_i = \sqrt{\mu \left[ \frac{2}{r_i} - \frac{1}{a} \right]}$$

WHERE  
 $r_i = 3397.8 \text{ km}$ 

$$h = r v \cos \beta$$

$$\beta = \cos^{-1} \left[ \frac{h}{r_{\text{impact}} v_{\text{impact}}} \right]$$

DESCENT



$$* \cos \textcircled{1} = \cos(28.5^\circ) \cos(33^\circ) \Rightarrow \textcircled{1} = 42.52^\circ$$

$$* \frac{\sin \textcircled{2}}{\sin(28.5^\circ)} = \frac{\sin(90^\circ)}{\sin(42.52^\circ)} \Rightarrow \textcircled{2} = 44.912^\circ$$

$$* \textcircled{3} = \textcircled{2} - 30^\circ = 44.912^\circ - 30^\circ \Rightarrow \textcircled{3} = 14.912^\circ$$

SOLVE FOR PLANE CHANGE ANGLE  $\alpha$

$$* \frac{\sin \alpha}{\sin(42.52^\circ)} = \frac{\sin \textcircled{3}}{\sin(\pi - \theta)} \quad \text{BUT } \sin(\pi - \theta) = \sin \theta$$

$$\sin \alpha = \frac{\sin(14.912^\circ) \sin(42.52^\circ)}{\sin \theta} = \frac{0.17391}{\sin \theta}$$

9. PLANE CHANGE ANGLE  
(deg)

$$\alpha = \sin^{-1} \left[ \frac{0.17391}{\sin \theta} \right]$$

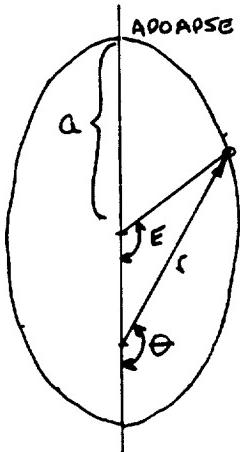
10. DE-ORBIT DELTA V.  
(km/sec)

$$\Delta V = \left[ v_{\text{circular}}^2 + v_{\text{apogee}}^2 - 2 v_{\text{circular}} v_{\text{apogee}} \cos \alpha \right]^{1/2}$$

$$v_{\text{circular}} = \sqrt{\frac{\mu}{r}} = 3.2714 \frac{\text{km}}{\text{sec}}$$

## DESCENT

TIME ELAPSED IN DEORBIT.



$E$  = ECCENTRIC ANOMALY, ANGLE  
MEASURED FROM CENTER OF ELLIPSE  
FROM PERIAPSE.

$\theta$  = TRUE ANOMALY, MEASURED FROM  
PERIAPSE AT MASS/FORCE CENTER

GIVEN  $\theta$  FIND TIME FROM DEORBIT  
AT APOAPSE:

$$E = \cos^{-1} \left[ e + \frac{r}{a} \cos \theta \right]$$

WHERE  $a$  IS THE SEMI-MAJOR AXIS AND  
 $r$  IS THE DISTANCE FROM THE MASS  
CENTER

$$t_r = \frac{E - e \sin E}{\sqrt{\mu/a^3}}$$

WHERE  $e$  IS ECCENTRICITY AND  $\mu$   
IS THE GRAVITATIONAL PARAMETER FOR  
MARS.

$$t_{apoapse} = \pi \sqrt{\frac{a^3}{\mu}}$$

$$\Delta t = t_{apoapse} - t_r$$

DESCENT  
FORCE  
BALANCE

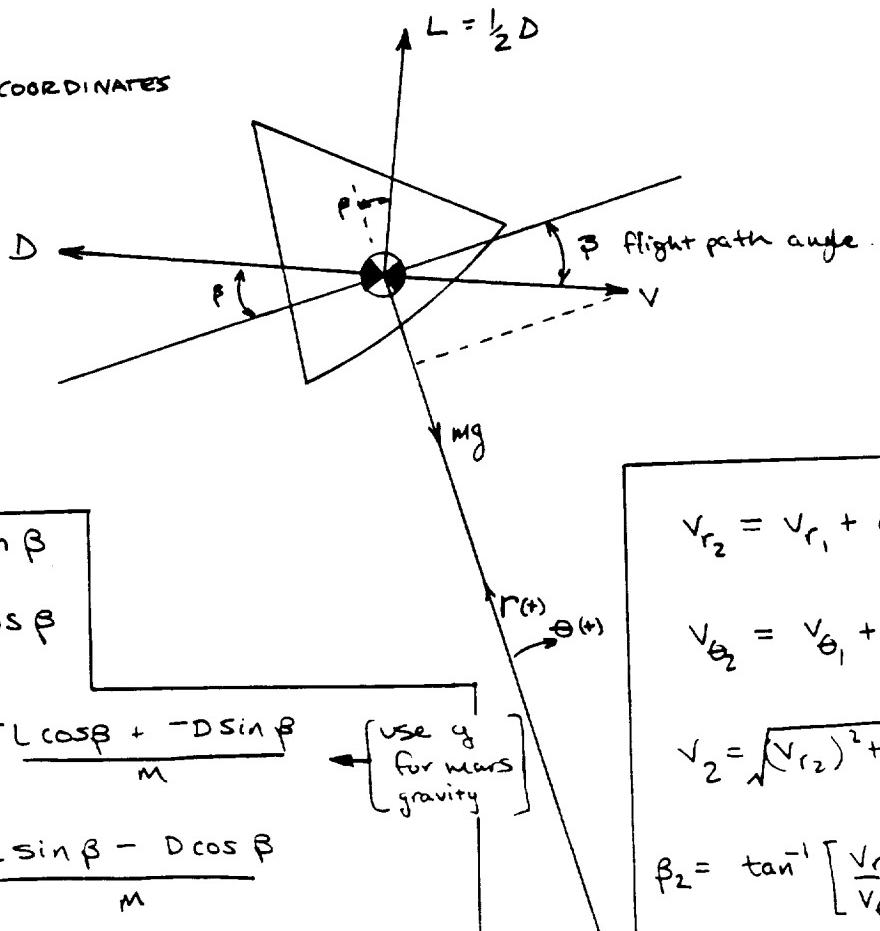
TAKE  $r$ ,  $v$ ,  $\beta$  FROM IDEAL ORBIT INITIALLY

CALCULATE INITIAL  $r$ ,  $\theta$ ,  $v_r$ ,  $v_\theta$   
 FROM THESE CALCULATE  $L$ ,  $D$

USE THESE TO  
 GENERATE NEXT  
 $r$ ,  $\theta$ ,  $v_r$ ,  $v_\theta$ ,  $L$ ,  $D$ , etc.

$L \neq D$  DEPENDS ON BODY (HEATSHIELD) AND/OR PARACHUTE

POLAR COORDINATES



$$v_r = v \sin \beta$$

$$v_\theta = v \cos \beta$$

$$a_r = g + \frac{-L \cos \beta - D \sin \beta}{m}$$

$$a_\theta = \frac{L \sin \beta - D \cos \beta}{m}$$

$$v_{r_2} = v_{r_1} + a_r \Delta t$$

$$v_{\theta_2} = v_{\theta_1} + a_\theta \Delta t$$

$$v_2 = \sqrt{(v_{r_2})^2 + (v_{\theta_2})^2}$$

$$\beta_2 = \tan^{-1} \left[ \frac{v_r}{v_\theta} \right]$$

$$r_2 = r_1 + \frac{(v_{r_2} + v_{r_1})}{2} \Delta t$$

$$\theta_2 = \theta_1 + \frac{(v_{\theta_2} + v_{\theta_1})}{2(r_2 + r_1)} \Delta t$$

1.	$r_1$	$\theta_1$	$v_{r_1}$	$v_{\theta_1}$	$v$	$\beta_1$	$L_1$	$D_1$	$a_r$	$a_\theta$
----	-------	------------	-----------	----------------	-----	-----------	-------	-------	-------	------------

2.	$r_2$	$\theta_2$	$v_{r_2}$	$v_{\theta_2}$	$v$	$\beta_2$	$L_2$	$D_2$	$a_{r_2}$	$a_{\theta_2}$
----	-------	------------	-----------	----------------	-----	-----------	-------	-------	-----------	----------------

3. CONTINUE TO "STEP DOWN" UNTIL  $r = r_{\text{surface}}$

## ORBITAL PARAMETERS:

GIVEN  $\epsilon$ 

ECCENTRICITY

$$1. \quad a = \frac{r_{\text{apogee}}}{[1 + e]}$$

SEMI-MAJOR AXIS

$$2. \quad p = a(1 - e^2)$$

ORBITAL PARAMETER

$$3. \quad h = \sqrt{p\mu}$$

ANGULAR MOMENTUM

## POSITION DEPENDENT BASIC VALUES

$$1. \quad \theta = \cos^{-1} \left[ \frac{p - r}{r_e} \right] \quad \text{TRUE ANOMALY AS FUNCT OF RADIUS}$$

$$2. \quad v = \sqrt{\mu \left[ \frac{2}{r} - \frac{1}{a} \right]} \quad \text{VELOCITY AS FUNCT OF RADIUS}$$

$$3. \quad \beta = \cos^{-1} \left[ \frac{h}{rv} \right] \quad \text{FLIGHT PATH ANGLE AS FUNCT OF RADIUS (AND VELOCITY WHICH IS A FUNCT. OF RADIUS)}$$

## DERIVED QUANTITIES NEEDED FOR ITERATION

$$1. \quad v_\theta = v \cos \beta \quad \text{TANGENTIAL VELOCITY AS FUNCT. OF } \beta \text{ AND VELOCITY}$$

$$2. \quad v_r = v \sin \beta \quad \text{RADIAL VELOCITY AS FUNCT. OF } \beta \text{ AND VELOCITY}$$

$$1. \quad g = \rho_0 e^{-\lambda r} \quad \text{DENSITY FUNCT. } (\rho_0 = \text{SURF. DENSITY}, \lambda = \text{SCALE HT.})$$

$$2. \quad g = \frac{\mu}{r^2} \quad \text{GRAVITATIONAL ACCEL AT GIVEN RADIUS}$$

IDEAL ORBITAL EQUATIONS  
DURING KEPLERIAN PHASE  
OF DESCENT

## ITERATIVE PROCEDURE

TAKE INITIAL VALUES FROM IDEAL ORBIT

RADIUS  $r$   
ANGLE  $\theta$   
VELOCITY  $v$   
COMPONENTS  $v_\theta, v_r$   
ANGLE  $\beta$

GIVEN  $m$  MASS  
GIVEN  $\mu$  GRAN. CONST.  
GIVEN  $r$  MARS MEAN  
VARIABLE  $m, A, C_D, K, \Delta t$

CALCULATE FORCES, ACCELERATIONS, VELOCITIES, POSITIONS

### 1. FORCES:

$$g = \frac{\mu}{r^2}$$

$$g = g_0 \exp\left[-\frac{r - r_{Mars}}{\lambda}\right]$$

$$D = \left[\frac{1}{2} g v^2\right] [A \ C_D]$$

$$L = K D$$

### 2. ACCELERATION

$$a_r = g - \left[ \frac{L \cos \beta + D \sin \beta}{m} \right]$$

$$a_\theta = \left[ \frac{L \sin \beta - D \cos \beta}{m} \right]$$

### 3. VELOCITIES

$$v_{r_{new}} = v_{r_{old}} + a_r \Delta t \quad v_{\theta_{new}} = \sqrt{v_{\theta_{new}}^2 + v_{r_{new}}^2}$$

$$v_{\theta_{new}} = v_{\theta_{old}} + a_\theta \Delta t$$

### 4. POSITIONS

$$\beta = \tan^{-1} \left[ \frac{v_{r_{new}}}{v_{\theta_{new}}} \right]$$

$$r_{new} = r_{old} + \frac{1}{2} [v_{r_{new}} + v_{r_{old}}] \Delta t$$

$$\theta_{new} = \theta_{old} + \left[ \frac{v_{\theta_{old}} + v_{\theta_{new}}}{r_{old} + r_{new}} \right] \Delta t$$

RETURN TO CALCULATE FORCES FOR NEXT STEP AND REPEAT CYCLE UNTIL LANDING.

FINAL  
DESCENT PROFILE

\* SEPARATION FROM MOV  $V = 3.272 \text{ km/sec}$  ALT = 600km

\*  $\Delta V \sim 8.0 \text{ m/s}$  FOR POSITIONING (ALT  $\pm 20 \text{ km}$ ,  $180^\circ$  IN 6 DAYS, 9.73 HOURS)

$2.5^\circ/\text{ORBIT}$  ADVANCE/REGDE MASS BEFORE DEORBIT  $\approx 80 \text{ kg}$

0.758 kg FUEL REQD.

\* DE ORBIT  $\Delta V = 167.9 \text{ m/s}$  TRANSFER ECCENTRICITY  $e = 0.10$

$a_r = 0.33 g's \downarrow$	$\beta_{250} = 6.3^\circ$	$v = 3421. \text{ m/s}$	ALT. 250 km	time (min) 29.02
$a_r = 0.34 g's \downarrow$	$\beta_{100} = 17.6^\circ$	$v = 3562. \text{ m/s}$	ALT. 100 km	32.52
$a_r = 2.61 g's \uparrow$	$\beta_{30} = 21.6^\circ$	$v = 1975. \text{ m/s}$	ALT. 30 km	33.60
$a_r = 0.25 \uparrow$	$\beta_{10} = 36.5^\circ$	$v = 330 \text{ m/s}$	ALT. 10 km	34.35

BASED ON DESCENT MASS OF 84.7 kg  $C_D = 1.0$

DIA = 1.5 m

$\sqrt{\Delta V_{\text{DEORBIT}}} = v_{\text{circular}} [1 - \sqrt{1-e}]$  ✓ REDUCTION OF  $\Delta V$  BY 300 m/s SAVES 10 kg

\* DEPLOY PARACHUTE AT -3.0 km ALT. / DROP HEAT SHIELD  
1.2 km ABOVE SURFACE 21 SECONDS BEFORE IMPACT

$$v = 108.8 \text{ m/s} \quad M =$$

$a_r = 27.5 \text{ m/s}^2 \uparrow \sim 3g's$  DURING FIRST SECOND

$$\beta = 78.9^\circ$$

PURSUE 10 SEC

AT ALT -3.5 km

20 SEC TO IMPACT

$$v = 56.5 \text{ m/s}$$

$$a_r = 3.35 \text{ m/s}^2 \uparrow \sim 1/3 g$$

$$\beta = 82.1^\circ$$

$$C_D = 1.0 \\ \text{DIA} = 4.0 \text{ m}$$

BASED ON MASS = 75.7 kg

\* RETRO FIRE AT < 100 m ABOVE SURFACE  
THRUST  $\sim 21 \text{ m/s}^2$  RESULT  $< 2g's$

REQUIRES 53.5 m TRAJECTORY TO BRING V TO ZERO.  
FIRES FOR  $\sim 2.5$  SEC.

IMPACT 36.1 MINUTES AFTER DEORBIT

\* REMAIN ON SURFACE  $\sim 12$  HOURS TO PERFORM SAMPLING MISSION  
(12 HRS, 19 MIN)

Chart1

Mars Atmospheric Density ( $\text{kg/m}^3$ ) Based on 1982 Standard Atmosphere

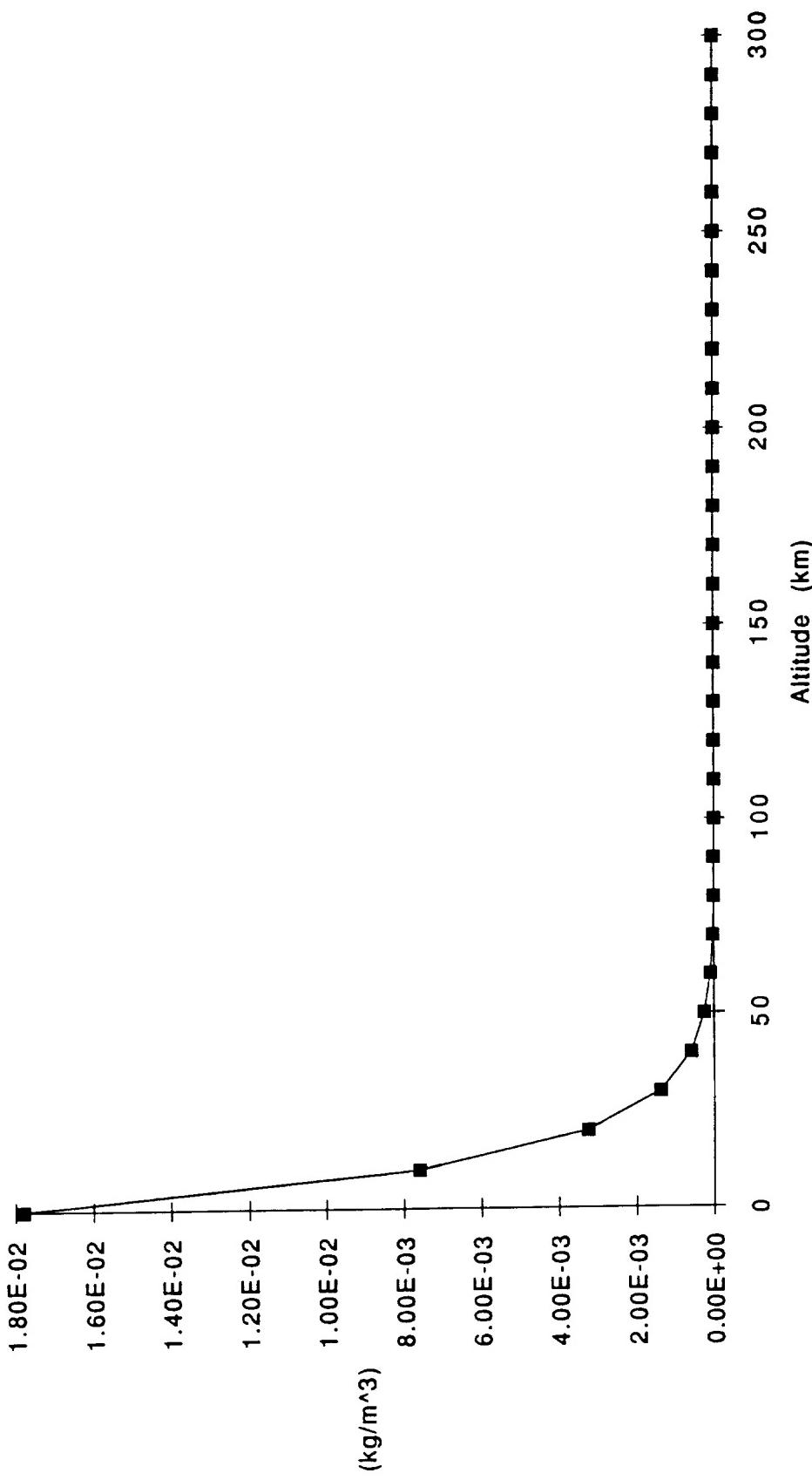
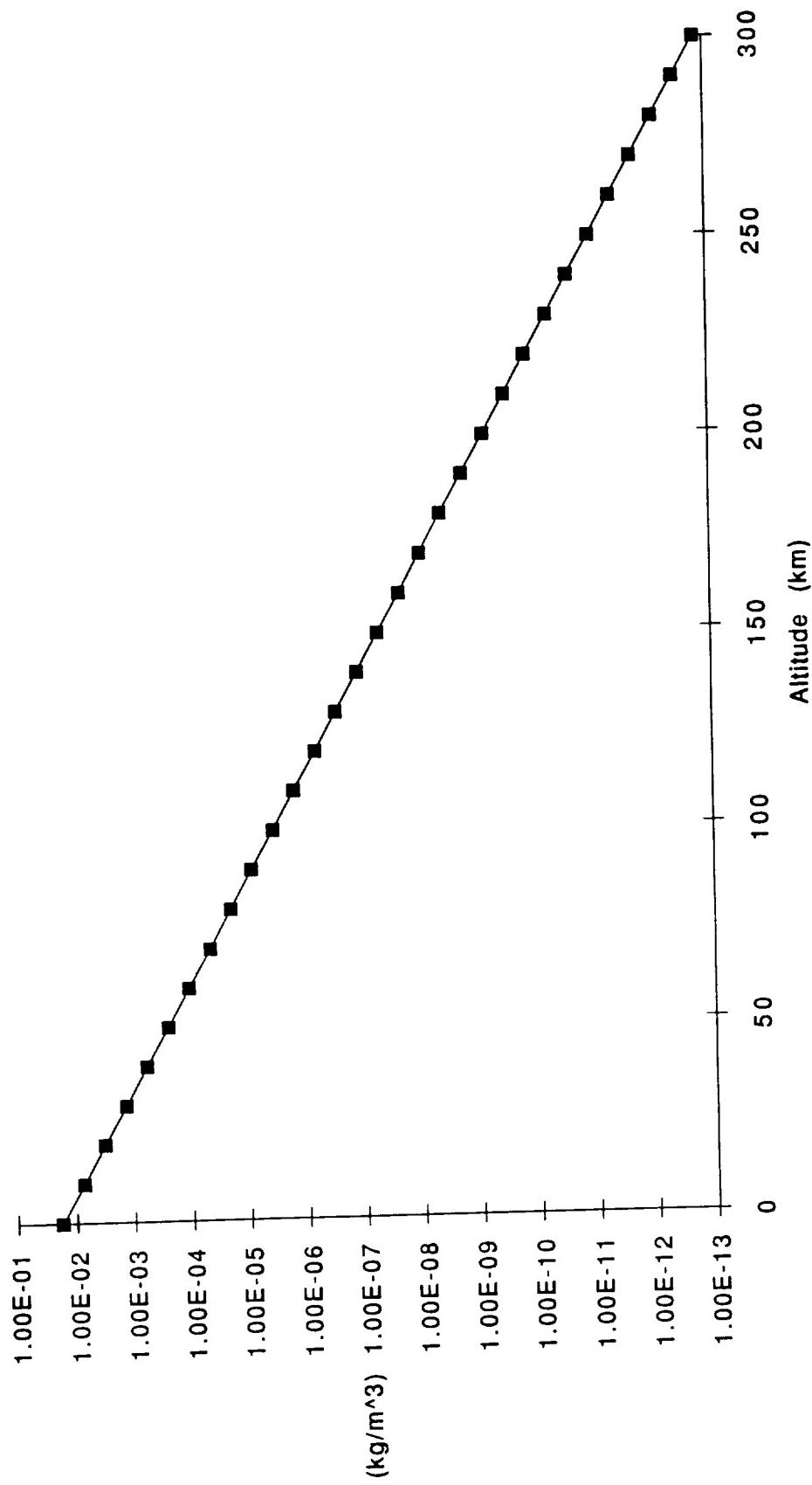


Chart1

Mars Atmospheric Density ( $\text{kg/m}^3$ ) Based on 1982 Standard Atmosphere



# HEAT SHIELD TERMINAL VELOCITY DESIGN CHART

VTERM\_MS.XLS Terminal Velocity

Alt (km)	Density (kg/m^3)	Sound(m/s)	D(m)	Terminal Velocity for Descent with Heat Shield							
				R(m)	Terminal Velocity (meters/second)				0.625	0.500	0.375
					1.250	1.125	1.000	0.875			
-4	1.35E-02	255		97.65	108.50	122.06	139.50	162.75	195.29	244.12	325.49
-2	1.18E-02	253		104.44	116.05	130.56	149.21	174.07	208.89	261.11	348.15
0	1.02E-02	252		112.34	124.82	140.42	160.48	187.23	224.68	280.84	374.46
2	8.89E-03	251		120.33	133.70	150.41	171.90	200.55	240.66	300.83	401.10
4	7.72E-03	249		129.13	143.47	161.41	184.47	215.21	258.25	322.82	430.42
6	6.67E-03	248		138.92	154.36	173.65	198.46	231.53	277.84	347.30	463.07
8	5.74E-03	247		149.75	166.39	187.19	213.93	249.59	299.50	374.38	499.17
10	4.91E-03	246		161.91	179.90	202.39	231.31	269.86	323.83	404.79	539.71
15	3.34E-03	244		196.31	218.13	245.39	280.45	327.19	392.63	490.79	654.38
20	2.28E-03	240		237.61	264.01	297.01	339.44	396.01	475.21	594.02	792.02
25	1.53E-03	236		290.06	322.20	362.57	414.36	483.43	580.11	725.14	966.85
30	9.99E-04	235		358.96	398.84	448.70	512.80	598.26	717.92	897.39	1196.53
35	6.49E-04	234		445.35	494.84	556.69	636.22	742.25	890.70	1111.38	1484.51
40	4.19E-04	233		554.27	615.85	692.83	791.81	923.78	1108.53	1385.67	1847.56
45	2.74E-04	230		685.41	761.57	856.76	979.16	1142.35	1370.82	1713.53	2284.70
50	1.76E-04	227		855.20	950.23	1069.01	1221.72	1425.34	1710.41	2138.01	2850.68
55	1.13E-04	222		1067.30	1185.89	1334.13	1524.72	1778.84	2134.60	2668.25	3557.67
60	7.12E-05	217		1344.58	1493.98	1680.72	1920.83	2240.96	2689.16	3361.45	4481.93
65	4.43E-05	212		1704.61	1894.01	2130.76	2435.15	2841.01	3409.21	4261.52	5682.02
70	2.72E-05	207		2175.41	2417.13	2719.27	3107.73	3625.69	4350.83	5438.53	7251.38
75	1.64E-05	203		2801.59	3112.88	3501.98	4002.27	4669.31	5603.18	7003.97	9338.63
80	9.75E-06	199		3633.49	4037.21	4541.86	5190.70	6055.82	7266.98	9083.72	12111.63
85	5.74E-06	195		4735.55	5261.72	5919.43	6765.07	7892.58	9471.10	11838.87	15785.16
90	3.32E-06	190		6226.69	6918.54	7783.36	8895.27	10377.81	12453.38	15566.72	20755.63
95	1.88E-06	186		8274.60	9194.00	10343.26	11820.86	13791.01	16549.31	20686.51	27582.01
100	1.02E-06	184		11233.78	12481.98	14042.23	16048.26	18722.97	22467.57	28084.46	37445.95
110	2.75E-07	206		21635.15	24039.05	27043.93	30907.35	36058.58	43270.29	54087.86	72117.15
120	8.72E-08	236		38420.93	42689.93	48026.17	54887.05	64043.89	76841.87	96052.34	128069.78
130	3.52E-08	252		60472.07	67191.19	75590.09	86388.67	100786.78	120944.14	151180.18	201573.57
140	1.61E-08	264		89415.59	99350.65	111769.48	127736.55	149025.98	178831.17	223538.97	298051.96
150	8.11E-09	272		125984.10	139982.33	157480.13	179977.29	209973.50	251968.20	314960.25	419497.00
160	4.34E-09	277		172219.10	191354.55	215273.87	246027.28	287031.83	344438.19	430547.74	574063.65
170	2.41E-09	281		231109.42	256788.24	288886.78	330156.31	385182.37	462218.84	577773.55	770364.73
180	1.34E-09	288		309937.25	344374.72	387421.56	442767.49	516562.08	619874.49	774843.12	1033124.15
190	7.86E-10	291		404682.94	449647.71	505853.68	578118.49	674471.57	809365.88	1011707.35	1348943.13
200	4.66E-10	294		525573.20	583970.22	656966.50	750818.86	875955.33	1051146.40	1313933.00	1751910.67
210	2.81E-10	297		676819.73	752021.92	846024.66	966885.33	1128032.88	1353639.46	1692049.32	2256065.76
220	1.71E-10	301		867617.12	964019.03	1084521.40	1239453.03	1446028.54	1735234.25	2169042.81	2892057.08
230	1.06E-10	305		1101978.79	1224420.88	1377473.49	1574255.42	1836631.32	2203957.59	2754946.98	3673262.64
240	6.60E-11	310		1396542.64	1551714.05	1745678.31	1995060.92	2327571.07	2793085.29	3491356.61	4655142.15
250	4.17E-11	315		1756944.90	1952161.00	2196181.13	2509921.29	2928241.50	3513889.80	4392362.25	5856483.01
260	2.66E-11	324		2199810.91	2444234.34	2749763.64	3142587.01	3666351.51	4399621.82	5499527.27	7332703.03
270	1.73E-11	332		2727740.91	3030821.24	3409676.14	3896772.74	4546234.86	5455481.83	6819352.29	9092469.72
280	1.14E-11	343		3360266.67	3733629.63	4200333.33	4800380.95	5600444.44	6720533.33	8400666.67	11200888.89
290	7.63E-12	354		4107370.70	4563745.22	5134213.37	5867672.43	6845617.83	8214741.40	10268426.75	13691235.66
300	5.22E-12	367		4965819.37	5517577.08	6207274.21	7094027.67	8276365.61	9931638.74	12414548.42	16552731.23
310	3.65E-12	380		5938540.70	6598378.56	7423175.88	8483639.57	9897567.83	11877081.40	14846351.75	19795135.67
320	2.61E-12	394		7022729.10	7803032.33	8778411.37	10032470.14	11704548.50	14045458.20	17556822.75	23409097.00
330	1.91E-12	407		8209363.50	9121515.00	10261704.37	11727664.14	13682272.49	16418726.99	20523408.74	27364544.99
340	1.43E-12	419		9487638.98	10541821.09	11859548.73	13553769.97	15812731.64	18975277.96	23719097.45	31625463.27
350	1.09E-12	430		10867081.41	12074534.90	13583851.76	15524402.01	18111802.35	21734162.82	27167703.53	36223604.70
360	8.52E-13	439		12291542.08	13657268.97	15364427.60	17559345.82	20485903.46	24583084.15	30728855.19	40971806.92
370	6.74E-13	448		13819623.82	15355137.57	17274529.77	19742319.74	23032706.36	27639247.63	34549059.54	46065412.72
380	5.42E-13	454		15410847.62	17123164.02	19263559.53	22015496.60	25684746.04	30821695.24	38527119.05	51369492.07
390	4.40E-13	460		17104084.42	19004538.24	21380105.52	24434406.31	28506807.36	34208168.83	42760211.04	57013614.72
400	3.61E-13	465		18883068.50	20981187.22	23603835.63	26975812.14	31471780.83	37766137.00	47207671.25	62943561.67
410	2.99E-13	468		20748684.90	23054094.34	25935856.13	29640978.43	34581141.50	41497369.81	51871712.26	69162283.01
420	2.49E-13	471		22736651.02	25262945.58	28420813.78	32480930.03	37894418.37	45473302.05	56841627.56	75788836.74
430	2.08E-13	473		24876799.28	27640888.09	31095999.10	35538284.69	41461332.14	49753598.56	62191998.20	82922664.27
440	1.75E-13	475		27121090.33	30134544.81	33901362.91	38744414.76	45201817.22	54242180.66	67802725.83	90403634.44
450	1.48E-13	477		29491394.72	32768216.35	36864243.40	42130563.88	49152324.53	58982789.44	73728486.79	98304649.06
460	1.25E-13	478		32090106.84	35655674.27	40112633.55	45843009.77	53483511.40	64180213.68	80225267.10	###
470	1.06E-13	479		34847629.20	38719588.00	43559536.50	49782327.43	58079382.00	69695258.40	87119072.99	###
480	9.03E-14	480		37755679.81	41950755.35	47194599.77	53936685.45	62926133.02	75511359.62	94389199.53	###

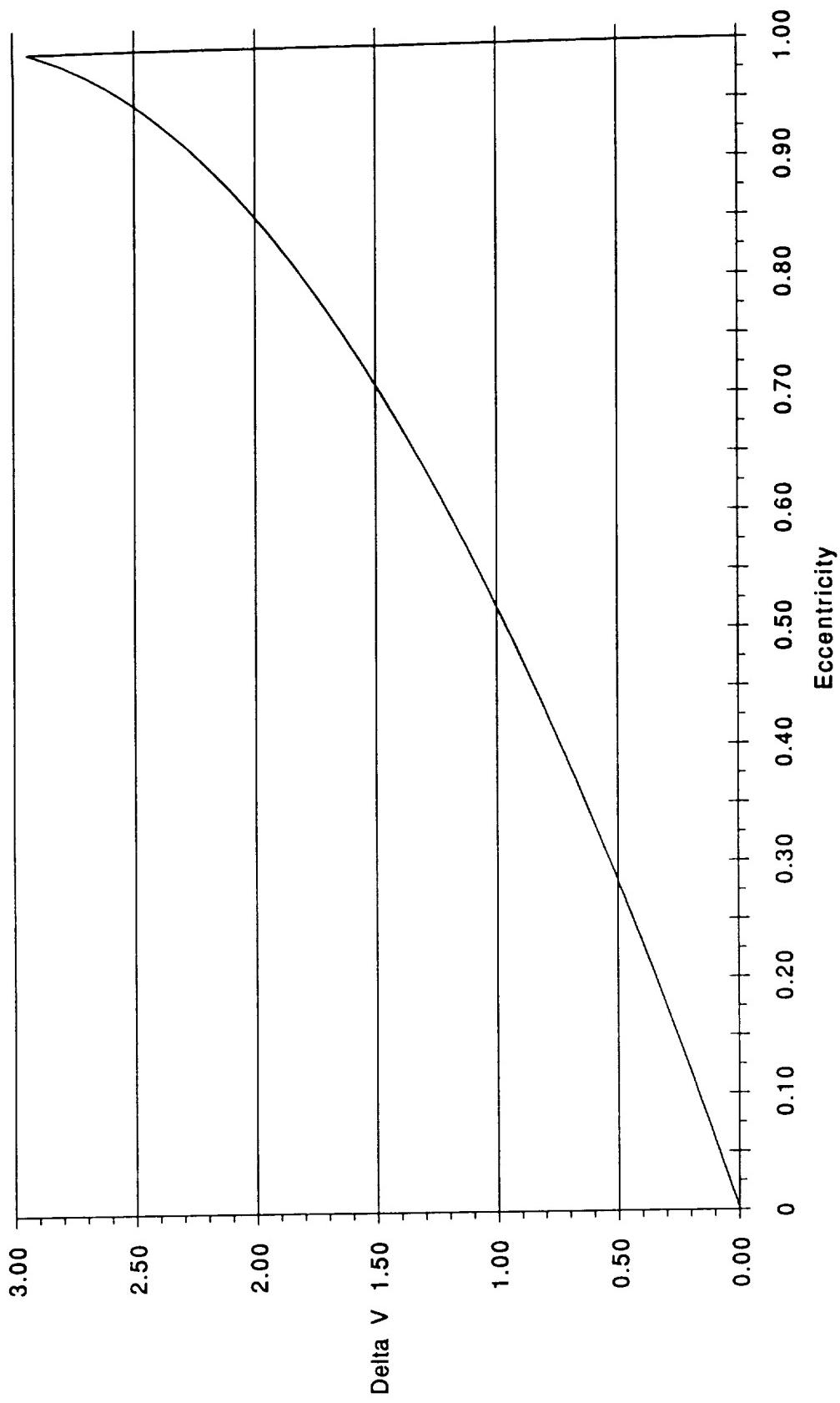
# PARACHUTE TERMINAL VELOCITY DESIGN CHART

VTERM\_PA.xls Terminal Velocity

	Weight (N)	282.36	Terminal Velocity for Descent with Parachute								
			Cd	1.00	Mass (kg)	75.70	Terminal Velocity (meters/second)				
		R(m)	4.500	4.000	3.500	3.000	2.500	2.000	1.500	1.000	
Alt (km)	Density (kg/m <sup>3</sup> )	Speed of Sound(m/s)	A(m <sup>2</sup> )	B(m)	C(m)	D(m)	E(m)	F(m)	G(m)	H(m)	
-4	1.35E-02	255		25.64	28.85	32.97	38.46	46.16	57.70	76.93	115.39
-2	1.18E-02	253		27.43	30.86	35.26	41.14	49.37	61.71	82.28	123.42
0	1.02E-02	252		29.50	33.19	37.93	44.25	53.10	66.38	88.80	132.75
2	8.89E-03	251		31.60	35.55	40.63	47.40	56.88	71.10	94.80	142.20
4	7.72E-03	249		33.91	38.15	43.60	50.86	61.04	76.30	101.73	152.59
6	6.67E-03	248		36.48	41.04	46.90	54.72	65.67	82.08	109.44	164.16
8	5.74E-03	247		39.33	44.24	50.56	58.99	70.79	88.48	117.98	176.96
10	4.91E-03	246		42.52	47.83	54.67	63.78	76.54	95.67	127.56	191.34
15	3.34E-03	244		51.55	58.00	66.28	77.33	92.80	115.99	154.66	231.99
20	2.28E-03	240		62.40	70.20	80.22	93.60	112.31	140.39	187.19	280.79
25	1.53E-03	236		76.17	85.69	97.93	114.26	137.11	171.38	228.51	342.77
30	9.99E-04	235		94.26	106.05	121.20	141.40	169.68	212.09	282.79	426.19
35	6.49E-04	234		116.95	131.57	150.37	175.43	210.51	263.14	350.86	526.28
40	4.19E-04	233		145.55	163.75	187.14	218.33	262.00	327.50	436.66	654.99
45	2.74E-04	230		179.99	202.49	231.42	269.99	323.99	404.98	539.98	809.97
50	1.76E-04	227		224.58	252.65	288.75	336.87	404.25	508.31	673.74	1010.62
55	1.13E-04	222		280.28	315.31	360.36	420.42	504.50	630.63	840.84	1261.26
60	7.12E-05	217		353.09	397.23	453.98	529.64	635.57	794.46	1059.28	1588.92
65	4.43E-05	212		447.64	503.59	575.54	671.46	805.75	1007.19	1342.92	2014.38
70	2.72E-05	207		571.28	642.68	734.50	856.91	1028.30	1285.37	1713.83	2570.74
75	1.64E-05	203		735.71	827.68	945.92	1103.57	1324.28	1655.35	2207.14	3310.71
80	9.75E-06	199		954.17	1073.45	1226.80	1431.26	1717.51	2146.89	2862.52	4293.78
85	5.74E-06	195		1243.58	1399.03	1598.89	1865.37	2238.45	2798.06	3730.74	5596.11
90	3.32E-06	190		1635.16	1839.56	2102.35	2452.74	2943.29	3679.12	4905.49	7358.23
95	1.88E-06	186		2172.96	2444.58	2793.80	3259.43	3911.32	4889.15	6518.87	9778.30
100	1.02E-06	184		2950.05	3318.81	3792.93	4425.08	5310.10	6637.62	8850.16	13275.24
110	2.75E-07	206		5681.51	6391.70	7304.79	8522.26	10226.71	12783.39	17044.52	25566.78
120	8.72E-08	236		10089.55	11350.74	12972.27	15134.32	18161.18	22701.48	30268.64	45402.96
130	3.52E-08	252		15880.30	17865.33	20417.52	23820.44	28584.83	35730.66	47640.89	71461.33
140	1.61E-08	264		23481.02	26416.15	30189.88	35221.53	42265.84	52832.30	70443.06	105664.59
150	6.11E-09	272		33084.11	37219.63	42336.71	49626.17	59551.40	74439.25	99252.33	148878.50
160	4.34E-09	277		45225.67	50878.88	58147.29	67838.51	81406.21	101757.76	136677.02	203815.53
170	2.41E-09	281		60690.59	68276.92	78030.76	91035.89	109243.07	136553.83	182071.78	273107.67
180	1.34E-09	288		81391.21	91565.11	104645.84	122086.81	146504.18	183130.22	244173.63	366260.44
190	7.86E-10	291		106271.95	119555.94	136635.36	159407.92	191289.50	239111.88	318815.84	478223.75
200	4.66E-10	294		138018.39	155207.68	177452.21	207027.58	248433.10	310541.37	414055.16	621082.74
210	2.81E-10	297		177736.55	199953.62	228518.42	266604.82	319925.79	399907.24	533209.65	799814.47
220	1.71E-10	301		227840.98	256321.11	292938.41	341761.48	410113.77	512642.21	683522.95	1025284.43
230	1.06E-10	305		289385.64	325558.84	372067.25	434078.45	502894.14	651117.68	868156.91	1302235.36
240	6.60E-11	310		366739.71	412582.17	471522.48	550109.56	660131.48	825164.34	1100219.13	1650328.69
250	4.17E-11	315		461383.31	519056.22	593207.11	692074.96	804869.95	1038112.44	1384149.92	2076224.87
260	2.66E-11	324		577682.33	649892.62	742734.43	866523.50	1039828.20	1299785.25	1733047.00	2599570.49
270	1.73E-11	332		716319.63	805859.58	920982.38	1074479.44	1289375.33	1611719.16	2148958.89	3223438.33
280	1.14E-11	343		882424.34	992727.38	1134545.58	1213836.51	1588363.81	1985454.76	2647273.01	3970909.52
290	7.63E-12	354		1078617.93	1213445.17	1386794.48	1617926.89	1941512.27	2426890.34	3235853.78	4853780.67
300	5.22E-12	367		1304051.23	1467057.63	1676637.29	1956076.84	2347292.21	2934115.26	3912153.68	5868230.52
310	3.65E-12	380		1559493.15	1754429.79	2005062.62	2339239.72	2807087.67	3508839.58	4678479.45	701771.17
320	2.61E-12	394		1844206.93	2074732.80	2371123.20	2766310.40	3319572.47	4149465.59	5512620.79	828931.19
330	1.91E-12	407		2155823.59	2425301.54	2771173.19	3233735.38	3880482.46	4850603.08	6467470.77	9701206.15
340	1.43E-12	419		2491505.70	2802943.91	3203364.47	3737258.55	4484710.26	5605887.82	7474517.09	11211775.64
350	1.09E-12	430		2853754.80	3210474.15	3669113.31	4280632.19	5136758.63	6420948.29	8561264.39	12841896.58
360	8.52E-13	439		3227825.93	3631304.17	4150061.91	4841738.89	5810086.67	7262608.34	9683477.79	14525216.68
370	6.74E-13	448		3629108.52	4082747.09	4665996.67	5443662.78	6532395.34	8165494.17	10887325.56	16330988.35
380	5.42E-13	454		4046972.56	4552844.13	5203250.44	6070458.85	7284550.61	9105688.27	12140917.69	18211376.54
390	4.40E-13	460		4491625.77	5053078.99	5774947.42	6737438.66	8084926.39	10106157.99	13474877.32	20212315.98
400	3.61E-13	465		4958796.69	5578646.27	6375595.74	7438195.03	8925834.04	11157292.55	14876390.06	22314585.09
410	2.99E-13	468		5448771.72	6129807.44	7005494.21	8173076.58	9807691.90	12259614.87	16346153.16	24519229.74
420	2.49E-13	471		5970768.46	6717114.51	7676702.30	8956152.69	10747383.22	13434229.03	17912305.37	26868458.06
430	2.08E-13	473		6532783.05	7349380.93	8399292.50	9799174.58	11759009.49	14698761.87	19598349.16	29397523.73
440	1.75E-13	475		7122146.11	8012414.38	9157045.00	10683219.17	12819863.00	16024828.75	21366438.33	32049657.50
450	1.48E-13	477		7744600.96	8712676.08	9957344.09	11616901.44	13940281.73	17425352.16	23233802.88	34850704.32
460	1.25E-13	478		8427036.92	9480416.54	10834761.76	12640555.39	15168666.46	18960833.08	25281110.77	37921666.16
470	1.06E-13	479		9151177.32	10295074.49	11765799.41	13726765.98	16472119.18	20590148.97	27453531.97	41180297.95
480	9.03E-14	480		9914847.26	11154203.17	12747660.76	14872270.89	17846725.07	22308406.33	29744541.78	44616812.67

Deorbit Velocity Change for Mars Descent

Deorbit Del V (km/sec) vs. Eccentricity



5/29/90 5:40 PM

## DELTA V DEORBIT VS ECCENTRICITY

$u_{\text{Mars}}$	42830.00	km <sup>3</sup> /sec <sup>2</sup>	Delta V = V circular • [ 1 - Sqrt( 1 - e ) ]		
Rapogee	4002.00	km			
V circular	3.27	km/sec			
e					

e	Del V (km/sec)								
0.01	0.02	0.21	0.36	0.41	0.76	0.61	1.23	0.81	1.85
0.02	0.03	0.22	0.38	0.42	0.78	0.62	1.25	0.82	1.88
0.03	0.05	0.23	0.40	0.43	0.80	0.63	1.28	0.83	1.92
0.04	0.07	0.24	0.42	0.44	0.82	0.64	1.31	0.84	1.96
0.05	0.08	0.25	0.44	0.45	0.85	0.65	1.34	0.85	2.00
0.06	0.10	0.26	0.46	0.46	0.87	0.66	1.36	0.86	2.05
0.07	0.12	0.27	0.48	0.47	0.89	0.67	1.39	0.87	2.09
0.08	0.13	0.28	0.50	0.48	0.91	0.68	1.42	0.88	2.14
0.09	0.15	0.29	0.51	0.49	0.94	0.69	1.45	0.89	2.19
0.10	0.17	0.30	0.53	0.50	0.96	0.70	1.48	0.90	2.24
0.11	0.19	0.31	0.55	0.51	0.98	0.71	1.51	0.91	2.29
0.12	0.20	0.32	0.57	0.52	1.00	0.72	1.54	0.92	2.35
0.13	0.22	0.33	0.59	0.53	1.03	0.73	1.57	0.93	2.41
0.14	0.24	0.34	0.61	0.54	1.05	0.74	1.60	0.94	2.47
0.15	0.26	0.35	0.63	0.55	1.08	0.75	1.64	0.95	2.54
0.16	0.27	0.36	0.65	0.56	1.10	0.76	1.67	0.96	2.62
0.17	0.29	0.37	0.67	0.57	1.13	0.77	1.70	0.97	2.70
0.18	0.31	0.38	0.70	0.58	1.15	0.78	1.74	0.98	2.81
0.19	0.33	0.39	0.72	0.59	1.18	0.79	1.77	0.99	2.94
0.20	0.35	0.40	0.74	0.60	1.20	0.80	1.81	1.00	#NUM!

**FINAL DESCENT PROFILE**

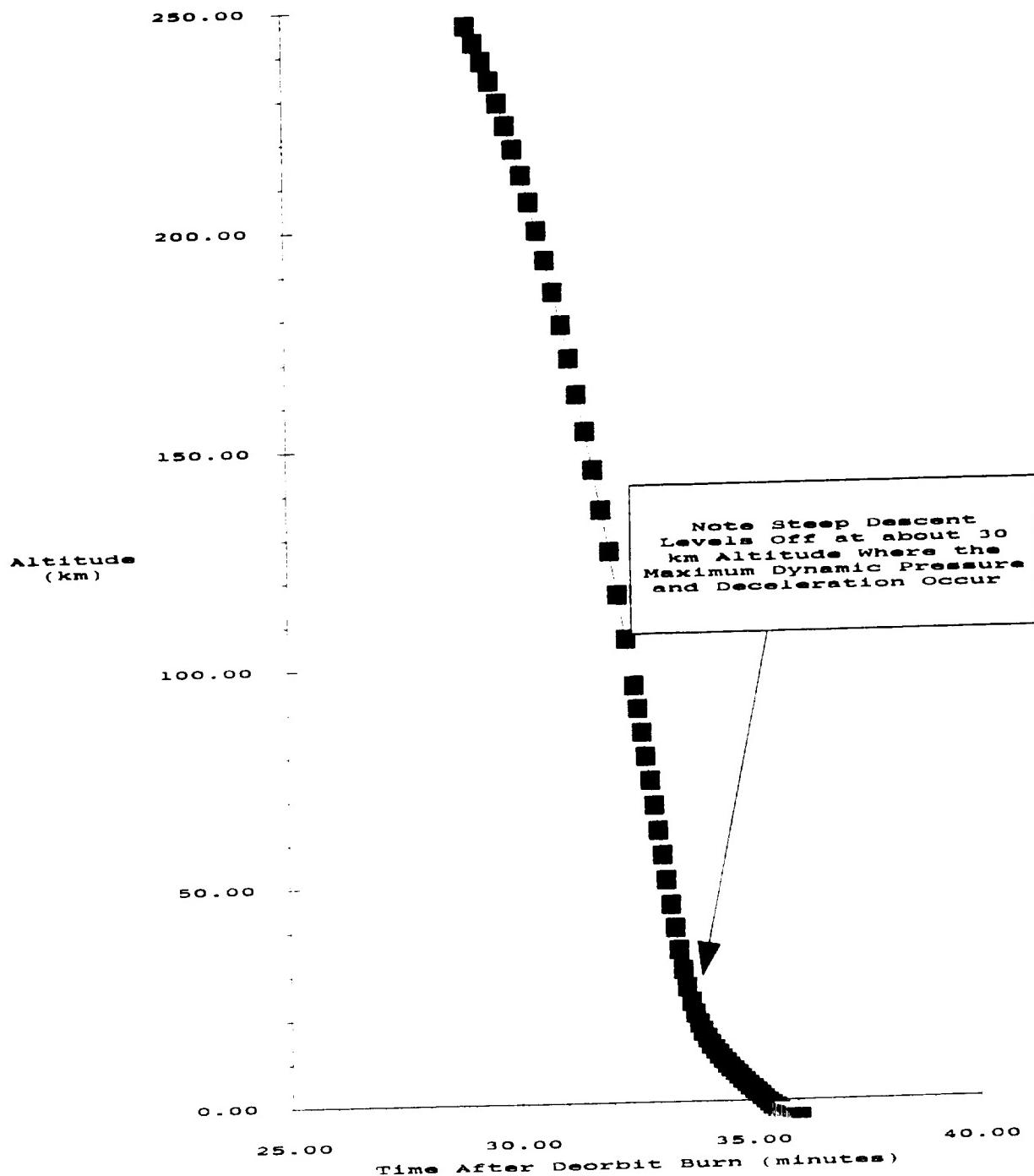
DESCO10.xls Incremental Descent in Actual Atmosphere (Eccentricity 0.10)

	A	B	C	D	E	F	G	H	I	J	K	L	M
1	W Mass:	42830	km/sec2	4.283E+13	m/sec2								
2	Repose:	4002	km	4003000	m								
3	R mean:	3402	km	3402000	m								
4	R elite:	3397.8	km	3397800	m								
5	V circular:	3.27214	km/sec	3272.1414	m/sec								
6	Rho m:	0.0178	kg/m3										
7	Lambda:	11.75	km										
8	Cd:	1.0											
9													
10	Altitude	m	A	Del T	g	Rho	Drag	at	st	vt new	vt new	v new	Beta new
11	(km)	(kg)	(m2)	(sec)	(m/sec2)	(kg/m3)	(N)	(m/sec2)	(m/sec2)	(m/sec)	(m/sec)	(m/sec)	(rad)
12	250.0	84.7	1.767	10.0	3.22	1.048E-11	3.21E+00	-1.24E+00	373.68	3400.97	3431.44	0.109	
13	246.4	84.7	1.767	10.0	3.22	1.048E-11	1.44E+00	3.22E+00	405.65	3400.97	3429.10	0.119	
14	242.5	84.7	1.767	10.0	3.22	1.048E-11	2.00E+00	3.22E+00	438.10	3400.97	3429.07	0.128	
15	238.1	84.7	1.767	10.0	3.22	2.77E-11	3.88E+00	3.23E+00	470.42	3400.97	3433.39	0.137	
16	233.8	84.7	1.767	10.0	3.24	4.08E-11	4.38E+00	3.24E+00	502.82	3400.97	3437.94	0.147	
17	228.9	84.7	1.767	10.0	3.25	6.17E-11	6.44E+00	3.25E+00	535.31	3400.97	3462.84	0.156	
18	223.7	84.7	1.767	10.0	3.26	9.59E-11	1.00E+02	3.26E+00	567.89	3400.97	3446.06	0.165	
19	218.2	84.7	1.767	10.0	3.27	1.53E-10	1.61E+02	3.27E+00	600.87	3400.97	3483.89	0.175	
20	212.3	84.7	1.767	10.0	3.28	2.52E-10	2.66E+02	3.28E+00	633.35	3400.97	3459.44	0.184	
21	206.2	84.7	1.767	10.0	3.29	4.26E-10	4.51E+02	3.29E+00	666.35	3400.97	3468.62	0.193	
22	199.7	84.7	1.767	10.0	3.30	7.41E-10	7.87E+02	3.30E+00	699.37	3400.97	3473.11	0.203	
23	193.0	84.7	1.767	10.0	3.31	1.33E-09	1.41E+02	3.31E+00	732.41	3400.97	3478.94	0.212	
24	185.7	84.7	1.767	10.0	3.32	2.44E-09	2.61E+02	3.33E+00	765.49	3400.97	3488.09	0.221	
25	178.3	84.7	1.767	10.0	3.34	4.61E-09	4.95E+02	3.34E+00	799.10	3400.96	3493.88	0.231	
26	170.4	84.7	1.767	10.0	3.36	8.97E-09	9.48E+02	3.36E+00	833.66	3400.95	3501.40	0.240	
27	162.2	84.7	1.767	10.0	3.37	1.60E-08	1.93E+01	3.37E+00	866.37	3400.95	3509.84	0.249	
28	155.7	84.7	1.767	10.0	3.39	3.70E-08	4.02E+01	3.38E+00	900.33	3400.88	3518.01	0.259	
29	144.9	84.7	1.767	10.0	3.40	7.85E-08	8.59E+01	3.40E+00	934.38	3400.78	3526.77	0.268	
30	135.7	84.7	1.767	10.0	3.42	1.71E-07	1.88E+00	3.42E+00	968.41	3400.57	3538.77	0.277	
31	126.2	84.7	1.767	10.0	3.44	3.65E-07	4.25E+00	3.43E+00	1003.68	3400.08	3544.88	0.287	
32	116.4	84.7	1.767	10.0	3.46	8.91E-07	9.89E+00	3.43E+00	1036.98	3398.96	3553.82	0.296	
33	106.2	84.7	1.767	10.0	3.48	1.13E-06	2.37E+01	3.46E+00	1070.94	3396.39	3561.14	0.305	
34	95.6	84.7	1.767	5.0	3.50	3.20E-06	5.83E+01	3.29E+00	1087.41	3393.01	3563.00	0.310	
35	90.2	84.7	1.767	5.0	3.51	8.24E-06	9.24E+01	3.18E+00	1103.30	3387.82	3562.94	0.315	
36	84.7	84.7	1.767	5.0	3.52	1.31E-05	1.47E+02	3.08E+00	1118.32	3379.59	3559.74	0.320	
37	79.2	84.7	1.767	5.0	3.53	2.11E-05	2.16E+02	2.68E+00	1131.52	3366.23	3551.41	0.324	
38	73.6	84.7	1.767	5.0	3.55	4.04E-05	3.79E+02	2.12E+00	1141.12	3349.13	3534.74	0.329	
39	67.9	84.7	1.767	5.0	3.56	5.51E-05	6.09E+02	1.23E+00	1148.30	3311.12	3504.89	0.334	
40	62.2	84.7	1.767	5.0	3.57	8.98E-05	9.74E+02	-2.00E+01	1147.30	3256.79	3492.97	0.339	
41	56.4	84.7	1.767	5.0	3.58	1.44E-04	1.54E+03	-2.47E+00	1144.98	3210.97	3367.97	0.344	
42	50.7	84.7	1.767	5.0	3.59	2.38E-04	3.89E+03	-2.20E+01	1082.41	3108.53	3233.40	0.349	
43	45.1	84.7	1.767	5.0	3.60	3.54E-04	1.07E+04	-3.92E+01	1053.15	2842.27	3030.77	0.355	
44	39.7	84.7	1.767	5.0	3.62	6.08E+03	4.92E+01	-5.65E+01	960.42	2549.94	2746.70	0.361	
45	34.7	84.7	1.767	5.0	3.63	9.32E-04	6.21E+03	-2.23E+01	856.13	2226.88	2386.47	0.368	
46	30.1	84.7	1.767	5.0	3.64	1.37E-03	6.92E+03	-2.57E+01	729.47	1848.79	1884.71	0.376	
47	26.1	84.7	1.767	5.0	3.64	1.93E-03	6.71E+03	-2.58E+01	602.16	1477.55	1599.84	0.387	
48	22.8	84.7	1.767	5.0	3.65	2.54E-03	5.75E+03	-2.20E+01	492.31	1162.96	1263.64	0.400	
49	20.1	84.7	1.767	5.0	3.66	3.23E-03	4.85E+03	-1.73E+01	405.81	919.60	1001.80	0.417	
50	17.8	84.7	1.767	5.0	3.66	3.91E-03	3.46E+03	-1.28E+01	341.35	728.64	804.89	0.438	
51	15.9	84.7	1.767	5.0	3.67	4.58E+03	2.62E+03	-9.48E+00	329.84	588.92	657.85	0.443	
52	14.4	84.7	1.767	5.0	3.67	5.29E-03	2.01E+03	-4.91E+00	319.40	482.89	547.89	0.449	
53	13.0	84.7	1.767	5.0	3.67	5.80E-03	1.57E+03	-5.08E+00	314.02	401.21	464.47	0.458	
54	11.7	84.7	1.767	5.0	3.68	6.55E-03	1.25E+03	-3.76E+00	313.22	337.50	400.39	0.468	
55	10.6	84.7	1.767	5.0	3.68	7.21E-03	1.02E+03	-2.80E+00	301.22	266.69	350.25	0.473	
56	9.6	84.7	1.767	5.0	3.68	7.88E-03	8.54E+02	-3.11E+00	180.65	245.43	310.77	0.480	
57	8.6	84.7	1.767	5.0	3.68	8.54E-03	7.31E+02	-1.61E+00	182.39	211.35	253.95	0.487	
58	7.8	84.7	1.767	5.0	3.68	9.21E-03	6.39E+02	-1.75E+00	171.39	158.96	233.49	0.494	
59	6.8	84.7	1.767	5.0	3.69	1.02E-02	5.71E+02	-9.41E+00	167.34	137.75	216.74	0.498	
60	5.8	84.7	1.767	5.0	3.69	1.18E-02	5.19E+02	-8.11E+00	163.87	119.71	202.94	0.504	
61	5.0	84.7	1.767	5.0	3.69	1.6E-02	4.81E+02	-6.93E+00	160.77	103.96	191.46	0.507	
62	4.2	84.7	1.767	5.0	3.69	2.14E-02	4.52E+02	-6.10E+00	157.86	90.14	181.78	0.502	
63	3.4	84.7	1.767	5.0	3.69	3.11E-02	4.3LE+02	-5.03E+00	155.00	77.94	173.50	0.495	
64	2.6	84.7	1.767	5.0	3.69	4.15E-02	4.15E+02	-5.71E+00	152.12	67.21	166.31	0.485	
65	1.8	84.7	1.767	5.0	3.70	5.22E-02	4.05E+02	-6.27E+00	149.15	57.73	159.94	0.473	
66	1.1	84.7	1.767	5.0	3.70	6.43E-02	3.97E+02	-1.67E+00	144.06	49.38	154.20	0.455	
67	0.3	84.7	1.767	5.0	3.70	7.92E+01	6.14E+01	-1.67E+00	142.90	42.05	148.96	0.425	
68	-0.4	84.7	1.767	5.0	3.70	8.57E+01	3.88E+02	-1.28E+00	139.62	35.63	144.09	0.321	
69	-1.2	84.7	1.767	5.0	3.70	9.96E+01	3.54E+02	-2.75E+01	136.27	30.05	139.54	0.354	
70	-1.9	84.7	1.767	5.0	3.70	1.28E+02	3.82E+02	-6.71E+01	136.72	20.90	108.75	0.377	
71	-2.5	84.7	1.767	5.0	3.71	2.11E+02	3.80E+02	-6.79E+01	135.59	29.08	138.67	0.359	
72	-2.7	84.7	1.767	5.0	3.71	3.24E+02	3.80E+02	-6.81E+01	134.22	27.23	136.96	0.371	
73	-2.8	84.7	1.767	5.0	3.71	4.26E+02	3.80E+02	-6.83E+01	130.72	20.90	108.75	0.377	
74	-3.0	84.7	12.566	1.0	3.71	5.20E+02	3.70E+03	-6.75E+01	106.72	17.00	92.11	1.385	
75	-3.1	84.7	12.566	1.0	3.71	6.31E+02	3.72E+03	-6.62E+01	90.52	13.97	70.36	1.394	
76	-3.2	84.7	12.566	1.0	3.71	7.32E+02	3.74E+03	-6.48E+01	78.10	13.97	64.00	1.403	
77	-3.3	84.7	12.566	1.0	3.71	8.35E+02	7.43E+02	-5.96E+00	69.73	11.81	60.72	1.403	
78	-3.3	75.7	12.566	1.0	3.71	9.36E+02	7.43E+02	-4.41E+00	63.76	10.17	64.57	1.413	
79	-3.4	75.7	12.566	1.0	3.71	1.03E+02	9.35E+02	-1.06E+00	55.99	7.82	56.54	1.433	
80	-3.5	75.7	12.566	1.0	3.71	1.14E+02	9.35E+02	-2.60E+00	53.39	6.94	51.44	1.442	
81	-3.5	75.7	12.566	1.0	3.71	1.24E+02	9.35E+02	-6.48E+01	49.71	5.35	51.72	1.451	
82	-3.6	75.7	12.566	1.0	3.71	1.34E+02	9.35E+02	-1.07E+00	47.33	4.99	47.55	1.476	
83	-3.7	75.7	12.566	1.0	3.71	1.44E+02	9.35E+02	-4.92E+01	47.33	4.49	47.55	1.476	
84	-3.												

DESCO10.xls Incremental Descent in Actual Atmosphere (Eccentricity 0.10)

	N	O	P	Q	R	S	T	U	V	W	X	Y	Z
4	GIVEN												
5	Beta 1	R L	R I	Ale 1	Ale I	Thetaair)	DEL Theta	E	n	t approp	Time(r)	Del Tl	Del Ti
6	(deg.)	(m)	(km)	(m)	(km)	(rad.)	(deg.)	(deg.)	(Rad/sec)	(sec)	(sec)	(min)	(min)
7	9.7350871	3652000	3652	250000	250	1.708861886	82.09916816	1.608788	0.000094308	3331.21234	1508.93	1732.383	28.0947
8													
9													
10	Beta new	r new	r new	new alt	new alt	del theta new	t new	t new					
11	(deg.)	(m)	(km)	(m)	(km)	(deg.)	(sec)	(min)					
12	6.37	3448426	34484.6	344846	246.43	82.63	1741.3	29.02					
13	6.81	3448526	34484.8	242256	242.83	82.17	1751.3	29.19					
14	7.34	3448626	34484.3	238306	238.31	81.70	1761.3	29.36					
15	7.88	3448764	3438.8	232764	233.76	84.24	1771.3	29.53					
16	8.41	3450098	3830.9	228099	228.90	84.77	1781.3	29.60					
17	8.94	3452707	3829.7	223707	223.71	85.31	1791.3	29.68					
18	9.48	3452911	3826.3	218191	218.19	86.85	1801.3	30.02					
19	10.01	3451439	3814.3	213249	212.38	86.39	1811.3	30.19					
20	10.55	3450817	3806.2	206179	206.18	86.93	1821.3	30.35					
21	11.08	3450161	3801.7	199681	199.66	87.47	1831.3	30.52					
22	11.63	3450853	3949.6	192683	192.85	88.01	1841.3	30.69					
23	12.18	3587698	3587.7	185698	185.70	89.55	1851.3	30.85					
24	12.69	3580208	3580.3	178208	178.20	89.10	1861.3	31.02					
25	13.22	3572361	3573.4	170281	170.38	89.64	1871.3	31.19					
26	13.74	3544323	3564.4	162323	162.23	90.18	1881.3	31.35					
27	14.26	3555737	3555.7	153727	153.73	90.73	1891.3	31.52					
28	14.83	3546894	3548.6	144894	144.89	91.28	1901.3	31.69					
29	15.34	3537721	3537.7	138721	135.72	91.83	1911.3	31.85					
30	15.90	3528208	3528.3	132308	126.21	92.38	1921.3	32.02					
31	16.43	3518383	3518.4	136283	116.38	92.94	1931.3	32.19					
32	16.97	3505818	3505.2	106184	106.18	93.49	1941.3	32.35					
33	17.50	3497618	3497.6	95619	95.61	94.05	1951.3	32.52					
34	17.77	3493219	3493.3	90219	90.219	94.33	1956.3	32.60					
35	18.04	3486742	3486.7	84742	84.76	94.40	1961.3	32.69					
36	18.31	3481189	3481.1	79189	79.19	94.88	1966.3	32.77					
37	18.58	3475944	3475.6	73544	73.56	95.14	1971.3	32.85					
38	18.85	3469880	3469.9	67880	67.88	95.44	1976.3	32.94					
39	19.13	3464184	3464.3	62184	62.15	95.71	1981.3	33.02					
40	19.40	3458618	3458.6	56418	56.41	95.98	1986.3	33.10					
41	19.49	3452708	3452.7	50708	50.71	96.25	1991.3	33.19					
42	19.90	3447108	3447.1	45108	45.11	96.81	1996.3	33.27					
43	20.31	3442174	3441.1	39714	39.71	96.79	2001.3	33.35					
44	20.67	3436660	3436.7	36660	34.66	96.98	2006.3	33.44					
45	21.07	3432081	3432.1	30091	30.09	97.18	2011.3	33.52					
46	21.56	3426132	3426.1	26123	26.13	97.35	2016.3	33.60					
47	22.17	3424793	3424.8	22793	22.79	97.49	2021.3	33.69					
48	22.84	3422057	3422.1	20057	20.06	97.60	2026.3	33.77					
49	23.90	3419812	3419.8	17813	17.81	97.68	2031.3	33.85					
50	25.10	3417944	3417.9	15964	15.96	97.75	2036.3	33.94					
51	24.84	3416384	3416.4	14354	14.38	97.81	2041.3	34.02					
52	26.36	3414973	3415.0	12973	12.97	97.85	2046.3	34.10					
53	30.25	3412739	3413.7	11739	11.74	97.99	2051.3	34.19					
54	32.53	3412616	3412.6	10616	10.62	98.92	2056.3	34.27					
55	35.06	3411375	3411.6	9575	9.58	98.95	2061.3	34.35					
56	37.44	3410595	3410.6	8595	8.60	97.97	2066.3	34.44					
57	40.10	3409862	3409.7	7662	7.66	97.99	2071.3	34.52					
58	43.97	3408745	3408.8	6765	6.77	98.01	2076.3	34.60					
59	47.23	3407896	3407.9	5896	5.90	98.02	2081.3	34.69					
60	50.54	3407069	3407.0	5049	5.05	98.03	2086.3	34.77					
61	53.85	3406221	3406.2	4221	4.23	98.04	2091.3	34.85					
62	57.11	3405409	3405.4	3409	3.41	98.05	2096.3	34.94					
63	60.37	3404613	3404.6	2613	2.61	98.06	2101.3	35.02					
64	63.30	3403830	3403.8	1830	1.83	98.07	2106.3	35.10					
65	66.16	3403083	3403.1	1063	1.06	98.07	2111.3	35.19					
66	68.84	3402308	3402.3	309	0.31	98.08	2116.3	35.27					
67	71.32	3401571	3401.6	-429	-0.43	98.09	2121.3	35.35					
68	73.60	3400849	3400.8	-1151	-1.15	98.10	2126.3	35.44					
69	75.68	3400143	3400.1	-1857	-1.86	98.11	2131.3	35.52					
70	77.56	3399453	3399.5	-2547	-2.55	98.12	2136.3	35.60					
71	77.89	3399317	3399.3	-2663	-2.68	98.13	2137.3	35.63					
72	78.32	3399192	3399.2	-2818	-2.82	98.14	2138.3	35.64					
73	79.53	3399047	3399.0	-2955	-2.93	98.15	2139.3	35.65					
74	78.92	3398927	3398.9	-3073	-3.07	98.16	2140.3	35.67					
75	79.36	3398828	3398.8	-3172	-3.17	98.17	2141.3	35.69					
76	79.86	3398744	3398.7	-3256	-3.26	98.18	2142.3	35.70					
77	80.39	3398670	3398.7	-3330	-3.33	98.19	2143.3	35.72					
78	80.86	3398603	3398.6	-3397	-3.40	98.20	2144.3	35.74					
79	81.49	3398542	3398.5	-3458	-3.46	98.21	2145.3	35.75					
80	82.05	3398406	3398.5	-3516	-3.51	98.22	2146.3	35.77					
81	82.40	3398429	3398.4	-3571	-3.57	98.23	2147.3	35.79					
82	83.13	3398377	3398.4	-3623	-3.62	98.24	2148.3	35.80					
83	83.63	3398323	3398.3	-3674	-3.67	98.25	2149.3	35.82					

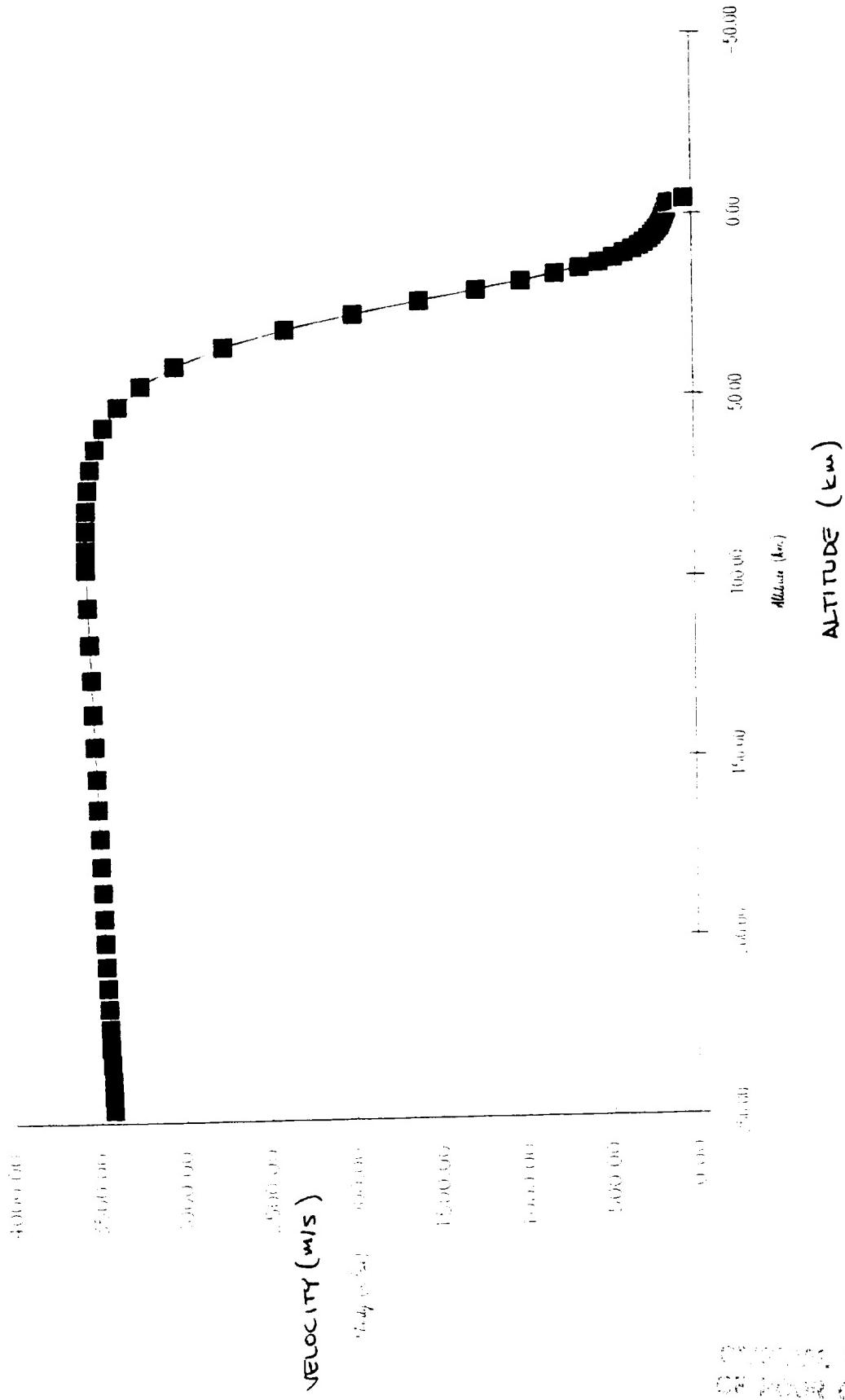
Altitude vs. Time for Descent with Eccentricity  
0.10 Atmospheric Phase



## DESCENT U

Model is altitude model with sharp steps of velocity 0.10

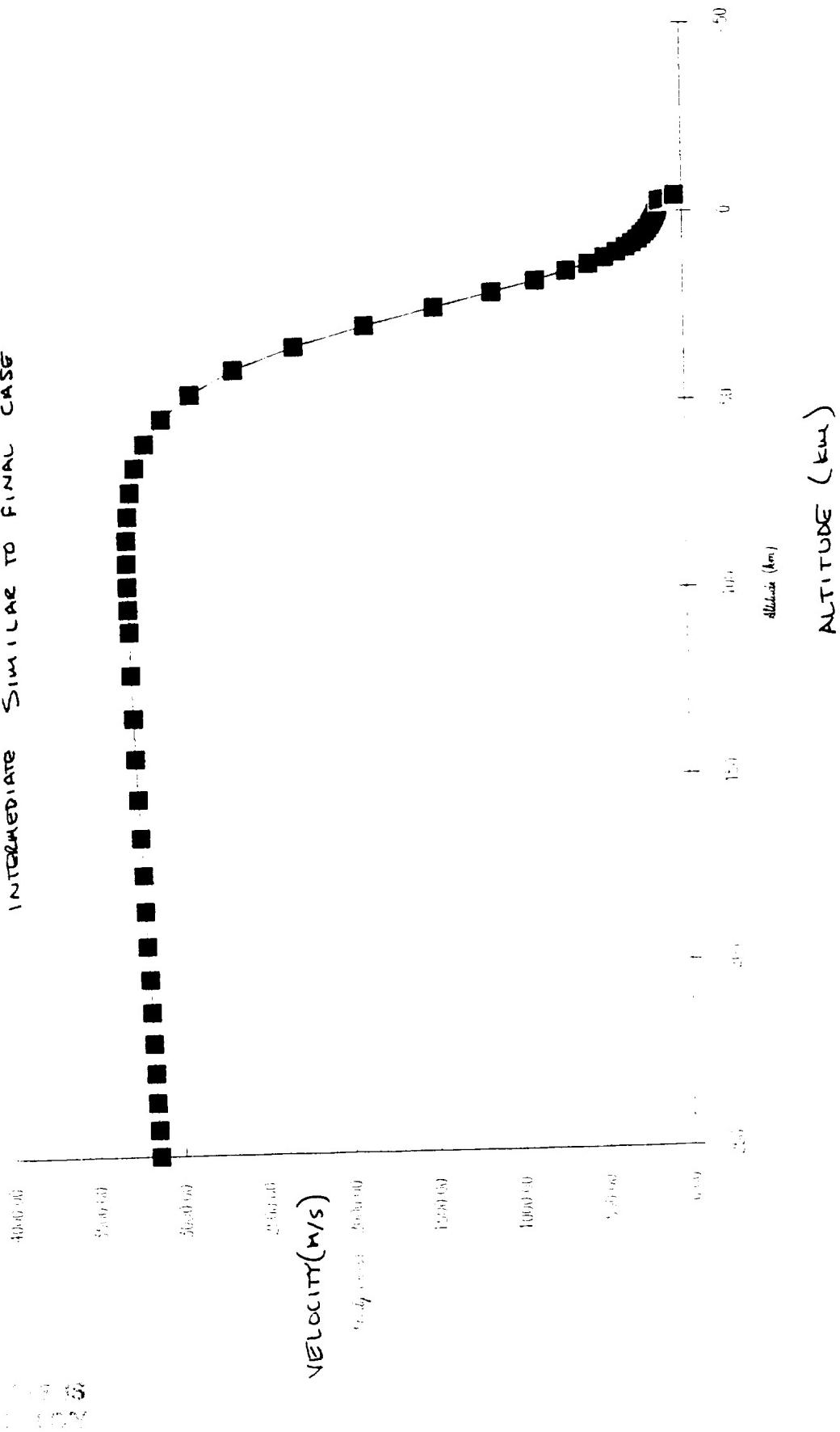
DESCENT U ECCENTRICITY = 0.10



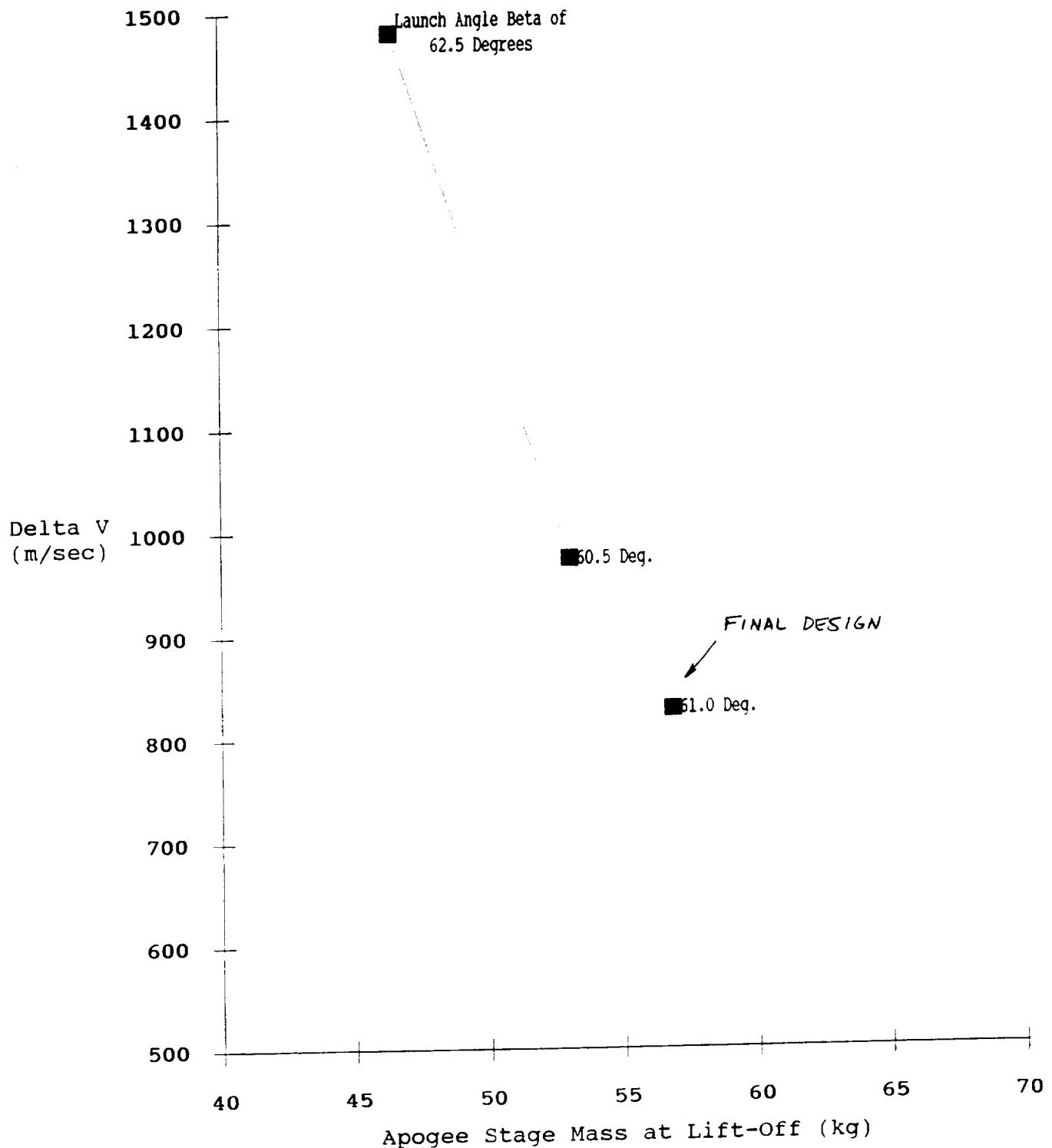
THE IMAGE IS  
OF POOR QUALITY

Study on Altitude Based with Sample Size with Eccentricity 0.27

DESCENT  $\downarrow$  ECCENTRICITY = 0.27  
INTERMEDIATE SIMILAR TO FINAL CASE



Minimum Insertion Delta V vs. Lift-Off Mass at Optimum Launch Angle



## DELVOPTM.XLS

56.8      Mass at Lift Off (kg)      14.9      Mass at Burn Out (kg)  
 133%      Of Original Fuel Mass (31.5 kg)  
 3271.414    V Circular

Beta LO (deg)	Max Alt (km)		Alt (km)	Beta (deg)	Velocity (m/sec)	Delta V (m/sec)
55	280					
60	541		600.91	2.92	2456.09	828.02
61	604		602.33	9.56	2470.65	930.44
61.5	636.21		601	19.37	2529.4	1219.57
63						

53      Mass at Lift Off (kg)      14.9      Mass at Burn Out (kg)  
 121%      Of Original Fuel Mass (31.5 kg)  
 3271.414    V Circular

Beta LO (deg)	Max Alt (km)		Alt (km)	Beta (deg)	Velocity (m/sec)	Delta V (m/sec)
59	517					
60	579					
60.5	609.82		601.18	4.96	2326.45	974.66
61	641.02		602.25	10.8	2345.89	1062.29

46.4      Mass at Lift Off (kg)      14.9      Mass at Burn Out (kg)  
 100%      Of Original Fuel Mass (31.5 kg)  
 3271.414    V Circular

Beta LO (deg)	Max Alt (km)		Alt (km)	Beta (deg)	Velocity (m/sec)	Delta V (m/sec)
61	530.31					
62	575.46		575.46	0.5	1805.89	1465.68
62.5	598.51		598.51	0	1790.72	1480.69
63	621.96		602.57	9.84	1801.06	1528.17

ASCENT61.XLS Ascent in Actual Atmosphere (Launch Angle 61 Degrees)

	A	B	C	D	E	F	G	H	I	J	K	L	M
1	u Mars:	42830	km3/sec2	4.263E+13	m3/sec2								
2	Rapogee	4002	km	4002000	m								
3	R mean	3402	km	3402000	m								
4	R site	3397.8	km	3397800	m								
5	V circular	3.2721414	km/sec	3272.1414	m/sec								
6	Rho m	0.0178	kg/m3										
7	Lambda	11.75	km										
8	Cd	1.0											
9													
10	Delta T	m	g	Rho	Drag	ar	at	vt new	vt new	v new	Beta	Beta	r
11	(sec)	(kg)	(m/sec2)	(kg/m3)	(N)	(m/sec2)	(m/sec2)	(m/sec)	(m/sec)	(m/sec)	(rad)	(deg.)	(m)
12	10.0	14.9	2.7	0.0	0.0	-2.7	0.0	125.1	2452.9	2456.1	0.1	2.9	4002906.5
13	Delta V Required for apogee:	828.74	Delta V at Launch (Thrust Only): 4020.32 Combined: 4849.06										
14													
15	Launch V	0	Area	0.101787602	Thrust	1500(N)							
16	Beta Launch	61	Cd	1									
17													
18													
19													
20	Delta T	m	g	Rho	Drag	ar	at	vt new	vt new	v new	Beta	Beta	r
21	(sec)	(kg)	(m/sec2)	(kg/m3)	(N)	(m/sec2)	(m/sec2)	(m/sec)	(m/sec)	(m/sec)	(rad)	(deg.)	(m)
22	1.0	56.0	3.71	2.54E-02	0.008E+00	1.94E+01	1.28E+01	19.39	12.80	23.23	1.065	61.00	3397800
23	1.0	56.3	3.71	2.54E-02	6.98E-01	1.96E+01	1.29E+01	38.97	25.72	46.69	0.988	56.58	3397819
24	1.0	55.6	3.71	2.53E-02	2.81E+00	1.87E+01	1.48E+01	57.66	40.50	70.46	0.959	54.92	3397858
25	1.0	55.3	3.71	2.52E-02	6.37E+00	1.84E+01	1.55E+01	76.07	56.03	94.47	0.936	53.63	3397916
26	1.0	54.8	3.71	2.50E-02	1.14E+01	1.82E+01	1.61E+01	94.24	72.15	118.69	0.917	52.57	3397992
27	1.0	54.3	3.71	2.48E-02	1.78E+01	1.80E+01	1.66E+01	112.23	88.75	143.08	0.902	51.66	3398086
28	1.0	53.8	3.71	2.46E-02	2.56E+01	1.78E+01	1.70E+01	130.03	105.77	167.61	0.888	50.88	3398199
29	1.0	53.2	3.71	2.43E-02	3.48E+01	1.76E+01	1.74E+01	147.67	123.13	192.27	0.876	50.18	3398239
30	1.0	52.7	3.71	2.40E-02	4.52E+01	1.75E+01	1.77E+01	165.15	140.80	217.02	0.865	49.55	3398476
31	1.0	52.2	3.71	2.37E-02	5.68E+01	1.73E+01	1.79E+01	182.47	158.72	241.05	0.855	48.98	3398641
32	1.0	51.7	3.71	2.33E-02	6.94E+01	1.72E+01	1.82E+01	199.63	176.88	266.72	0.846	48.46	3398824
33	1.0	51.2	3.71	2.29E-02	8.30E+01	1.70E+01	1.83E+01	216.64	195.23	291.62	0.837	47.98	3399244
34	1.0	50.7	3.71	2.25E-02	9.74E+01	1.68E+01	1.85E+01	233.48	213.74	316.54	0.829	47.53	3399474
35	1.0	50.2	3.71	2.21E-02	1.13E+02	1.67E+01	1.87E+01	250.16	232.41	341.46	0.822	47.11	3399724
36	1.0	49.7	3.71	2.16E-02	1.28E+02	1.65E+01	1.88E+01	266.68	251.20	366.36	0.815	46.71	3399990
37	1.0	49.2	3.71	2.11E-02	1.44E+02	1.64E+01	1.89E+01	283.04	270.10	391.24	0.809	46.34	3400274
38	1.0	48.7	3.70	2.06E-02	1.61E+02	1.62E+01	1.90E+01	299.25	289.10	416.09	0.803	45.99	3400573
39	1.0	48.2	3.70	2.01E-02	1.77E+02	1.61E+01	1.91E+01	315.30	308.19	440.90	0.797	45.65	3400888
40	1.0	47.7	3.70	1.96E-02	1.94E+02	1.59E+01	1.92E+01	331.20	327.35	465.67	0.791	45.34	3401219
41	1.0	47.1	3.70	1.90E-02	2.10E+02	1.58E+01	1.92E+01	346.96	346.58	490.41	0.786	45.03	3401566
42	1.0	46.6	3.70	1.85E-02	2.26E+02	1.56E+01	1.93E+01	363.58	365.89	515.11	0.781	44.74	3401929
43	1.0	46.1	3.70	1.79E-02	2.42E+02	1.55E+01	1.94E+01	378.08	385.26	539.79	0.776	44.46	3402307
44	1.0	45.6	3.70	1.73E-02	2.57E+02	1.54E+01	1.94E+01	393.46	404.70	564.44	0.771	44.19	3402700
45	1.0	45.1	3.70	1.68E-02	2.72E+02	1.53E+01	1.95E+01	408.74	424.22	589.09	0.767	43.93	3403109
46	1.0	44.6	3.70	1.62E-02	2.86E+02	1.52E+01	1.96E+01	423.92	443.82	613.75	0.762	43.69	3403533
47	1.0	44.1	3.70	1.56E-02	3.00E+02	1.51E+01	1.97E+01	439.03	463.51	638.42	0.758	43.45	3403972
48	1.0	43.6	3.70	1.50E-02	3.12E+02	1.50E+01	1.98E+01	454.07	483.29	663.13	0.754	43.21	3404426
49	1.0	43.1	3.70	1.45E-02	3.24E+02	1.50E+01	1.99E+01	469.06	503.18	687.90	0.750	42.99	3404895
50	1.0	42.6	3.69	1.39E-02	3.35E+02	1.50E+01	2.00E+01	484.03	523.20	712.75	0.747	42.77	3405379
51	1.0	42.1	3.69	1.34E-02	3.45E+02	1.49E+01	2.02E+01	498.98	543.35	737.70	0.743	42.56	3405878
52	1.0	41.6	3.69	1.28E-02	3.54E+02	1.50E+01	2.03E+01	513.93	563.65	762.78	0.739	42.36	3406392
53	1.0	41.1	3.69	1.22E-02	3.63E+02	1.50E+01	2.05E+01	528.90	584.12	788.00	0.736	42.16	3406921
54	1.0	40.5	3.69	1.17E-02	3.70E+02	1.50E+01	2.07E+01	543.92	604.78	813.40	0.732	41.97	3407465
55	1.0	40.0	3.69	1.12E-02	3.76E+02	1.51E+01	2.09E+01	559.00	625.65	839.00	0.729	41.78	3408024
56	1.0	39.5	3.69	1.07E-02	3.82E+02	1.52E+01	2.11E+01	574.16	646.75	864.84	0.726	41.60	3408598
57	1.0	39.0	3.69	1.02E-02	3.86E+02	1.53E+01	2.13E+01	589.42	668.09	890.93	0.723	41.42	3409187
58	1.0	38.5	3.69	9.66E-03	3.90E+02	1.54E+01	2.16E+01	604.80	689.70	917.32	0.720	41.25	3409792
59	1.0	38.0	3.68	9.17E-03	3.93E+02	1.55E+01	2.19E+01	620.33	711.61	944.03	0.717	41.08	3410413
60	1.0	37.5	3.68	8.70E-03	3.95E+02	1.57E+01	2.22E+01	636.02	731.83	971.10	0.714	40.92	3411049
61	1.0	37.0	3.68	8.24E-03	1.96E+02	1.59E+01	2.26E+01	651.90	756.40	998.56	0.711	40.76	3411701
62	1.0	36.5	3.68	7.80E-03	3.96E+02	1.61E+01	2.29E+01	667.99	779.33	1026.43	0.709	40.60	3412369
63	1.0	36.0	3.68	7.37E-03	3.95E+02	1.63E+01	2.33E+01	684.30	802.66	1054.77	0.706	40.45	3413053
64	1.0	35.5	3.68	6.95E-03	3.93E+02	1.66E+01	2.37E+01	700.87	826.41	1083.59	0.703	40.30	3413754
65	1.0	35.0	3.68	6.55E-03	3.91E+02	1.68E+01	2.42E+01	717.71	850.60	1112.94	0.701	40.16	3414471
66	1.0	34.4	3.67	6.16E-03	3.88E+02	1.71E+01	2.47E+01	734.85	875.27	1142.85	0.698	40.02	3415206
67	1.0	33.9	3.67	5.78E-03	3.85E+02	1.75E+01	2.52E+01	752.32	900.44	1173.36	0.696	39.88	3415959
68	1.0	33.4	3.67	5.43E-03	3.80E+02	1.78E+01	2.57E+01	770.12	926.15	1204.51	0.694	39.74	3416729
69	1.0	32.9	3.67	5.08E-03	3.75E+02	1.82E+01	2.63E+01	788.30	952.42	1238.33	0.691	39.61	3417517
70	1.0	32.4	3.67	4.75E-03	3.70E+02	1.86E+01	2.69E+01	806.87	979.28	1268.87	0.689	39.49	3418324
71	1.0	31.9	3.67	4.44E-03	3.64E+02	1.90E+01	2.75E+01	825.85	1006.77	1302.16	0.687	39.36	3418930
72	1.0	31.4	3.66	4.14E-03	3.57E+02	1.94E+01	2.81E+01	845.28	1034.92	1336.25	0.685	39.24	3419150
73	1.0	30.9	3.66	3.85E-03	3.50E+02	1.99E+01	2.88E+01	865.17	1061.76	1371.17	0.683	39.12	3419995
74	1.0	30.4	3.66	3.58E-03	3.42E+02	2.04E+01	2.96E+01	885.56	1093.33	1406.98	0.681	39.01	3420860
75	1.0	29.9	3.66	3.32E-03	3.34E+02	2.09E+01	3.03E+01	906.47	1123.66	1443.71	0.679	38.89	3421746
76	1.0	29.4	3.66	3.07E-03	3.26E+02	2.15E+01	3.11E+01	927.92	1154.79	1481.41	0.677	38.78	3422652
77	1.0	28.9	3.65	2.84E-03	3.17E+02	2.20E+01	3.20E+01	949.95	1186.75	1520.13	0.675	38.68	3423580
78	1.0	28.3	3.65	2.62E-03	3.08E+02	2.26E+01	3.28E+01	972.58	1219.59	1559.91	0.673	38.57	3424530
79	1.0	27.8	3.65	2.41E-03	2.98E+02	2.33E+01	3.37E+01	995.85	1253.33	1600.80	0.671	38.47	3425503
80	1.0	27.3	3.65	2.21E-03	2.89E+02	2.39E+01	3.47E+01	1019.77	1288.04	1642.85	0.670	38.37	3426499
81	1.0	26.8	3.65	2.03E-03	2.79E+02	2.46E+01</							

## ASCENT61.xls Ascent in Actual Atmosphere (Launch Angle 61 Degrees)

	A	B	C	D	E	F	G	H	I	J	K	L	M
91	1.0	21.7	3.62	7.51E-04	1.75E+03	3.35E+01	4.83E+01	1336.16	1745.90	2198.52	0.653	37.43	3439201
92	1.0	21.2	3.62	6.70E-04	1.65E+02	3.46E+01	4.99E+01	1370.75	1795.83	2259.20	0.652	37.35	3440537
93	1.0	20.7	3.62	5.96E-04	1.55E+02	3.58E+01	5.16E+01	1406.51	1847.42	2321.90	0.651	37.28	3441908
94	1.0	20.2	3.61	5.29E-04	1.45E+02	3.70E+01	5.33E+01	1443.49	1900.74	2386.73	0.650	37.21	3443314
95	1.0	19.7	3.61	4.68E-04	1.36E+02	3.83E+01	5.51E+01	1481.74	1955.86	2453.76	0.648	37.15	3444758
96	1.0	19.2	3.61	4.12E-04	1.26E+02	3.96E+01	5.70E+01	1521.33	2012.88	2523.12	0.647	37.08	3446239
97	1.0	18.7	3.60	3.62E-04	1.17E+02	4.10E+01	5.90E+01	1562.32	2071.88	2594.90	0.646	37.03	3447761
98	1.0	18.2	3.60	3.17E-04	1.09E+02	4.25E+01	6.11E+01	1604.78	2132.96	2669.24	0.645	36.96	3449323
99	1.0	17.7	3.60	2.77E-04	1.00E+02	4.40E+01	6.33E+01	1648.78	2196.22	2746.25	0.644	36.90	3450928
100	1.0	17.2	3.59	2.40E-04	9.23E+01	4.56E+01	6.56E+01	1694.41	2261.78	2826.07	0.643	36.84	3452577
101	1.0	16.7	3.59	2.08E-04	8.46E+01	4.73E+01	6.80E+01	1741.75	2329.76	2908.87	0.642	36.78	3454271
102	1.0	16.2	3.59	1.79E-04	7.73E+01	4.91E+01	7.05E+01	1790.90	2400.30	2994.79	0.641	36.73	3456013
103	1.0	15.6	3.58	1.54E-04	7.03E+01	5.11E+01	7.32E+01	1841.95	2473.53	3084.02	0.640	36.67	3457804
104	1.0	15.1	3.58	1.32E-04	6.38E+01	-6.09E+00	-3.38E+00	1835.86	2470.15	3077.67	0.639	36.63	3459646
105	1.0	14.9	3.57	1.13E-04	5.43E+01	-5.75E+00	-2.93E+00	1830.11	2467.23	3071.89	0.638	36.57	3461481
106	1.0	14.9	3.57	9.64E-05	4.63E+01	-5.42E+00	-2.50E+00	1824.69	2464.73	3066.66	0.637	36.51	3463312
107	1.0	14.9	3.57	8.26E-05	3.95E+01	-5.15E+00	-2.13E+00	1819.54	2462.60	3061.88	0.636	36.46	3465136
108	1.0	14.9	3.56	7.07E-05	3.37E+01	-4.91E+00	-1.82E+00	1814.63	2460.78	3057.50	0.635	36.41	3469556
109	5.0	14.9	3.54	3.27E-05	1.55E+01	-4.16E+00	-8.40E-01	1793.81	2456.58	3041.80	0.631	36.14	3476029
110	5.0	14.9	3.53	1.52E-05	7.17E+00	-3.81E+00	-3.89E-01	1774.76	2454.63	3029.03	0.626	35.87	3484998
111	5.0	14.9	3.51	7.16E-06	3.34E+00	-3.64E+00	-1.82E-01	1756.56	2453.73	3017.66	0.621	35.60	3493872
112	5.0	14.9	3.49	3.39E-06	1.57E+00	-3.55E+00	-8.57E-02	1738.80	2453.30	3007.01	0.617	35.33	3502655
113	5.0	14.9	3.47	1.62E-06	7.44E+00	-3.50E+00	-4.07E-02	1721.29	2453.09	2996.75	0.612	35.06	3511349
114	5.0	14.9	3.46	7.77E-07	3.55E+01	-3.47E+00	-1.93E-02	1703.93	2453.00	2986.73	0.607	34.79	3519955
115	5.0	14.9	3.44	3.76E-07	1.71E+01	-3.45E+00	-9.42E-03	1686.70	2452.95	2976.90	0.602	34.51	3528475
116	5.0	14.9	3.42	1.84E-07	8.28E+02	-3.43E+00	-4.58E-03	1669.57	2452.93	2967.20	0.598	34.24	3536908
117	5.0	14.9	3.41	9.03E-08	4.04E-02	-3.41E+00	-2.24E-03	1652.52	2452.91	2957.64	0.593	33.97	3545256
118	5.0	14.9	3.38	2.23E-08	9.85E+03	-3.38E+00	-5.50E-04	1618.67	2452.91	2938.05	0.583	33.42	3561696
119	5.0	14.9	3.36	1.12E-08	4.92E+03	-3.36E+00	-2.75E-04	1601.87	2452.90	2929.63	0.579	33.15	3569790
120	5.0	14.9	3.35	5.66E-09	2.47E+03	-3.35E+00	-1.39E-04	1585.14	2452.90	2920.51	0.574	32.87	3577799
121	5.0	14.9	3.33	2.88E-09	1.25E+03	-3.33E+00	-7.05E-05	1568.48	2452.90	2911.51	0.569	32.60	3585725
122	5.0	14.9	3.32	1.48E-09	6.38E+04	-3.32E+00	-3.61E-05	1551.90	2452.90	2902.61	0.564	32.32	3593567
123	5.0	14.9	3.30	7.64E-10	3.38E+04	-3.30E+00	-1.86E-05	1535.39	2452.90	2893.81	0.559	32.04	3601327
124	5.0	14.9	3.29	3.97E-10	1.69E+04	-3.29E+00	-9.64E-06	1518.95	2452.90	2885.12	0.554	31.77	3609004
125	5.0	14.9	3.27	2.08E-10	8.82E+05	-3.27E+00	-5.03E-06	1502.57	2452.90	2876.54	0.550	31.49	3616598
126	5.0	14.9	3.26	1.10E-10	4.63E+05	-3.26E+00	-2.65E-06	1486.27	2452.90	2868.05	0.545	31.21	3624111
127	5.0	14.9	3.25	5.84E-11	2.44E+05	-3.25E+00	-1.40E-06	1470.03	2452.90	2859.67	0.540	30.93	3631543
128	5.0	14.9	3.23	3.12E-11	1.30E+05	-3.23E+00	-7.48E-07	1453.86	2452.90	2851.39	0.535	30.66	3638893
129	5.0	14.9	3.23	1.66E-11	6.94E+06	-3.22E+00	-4.02E-07	1437.75	2452.90	2843.21	0.530	30.38	3646162
130	5.0	14.9	3.21	9.12E-12	3.75E+06	-3.21E+00	-2.17E-07	1421.70	2452.90	2835.13	0.525	30.10	3653351
131	5.0	14.9	3.20	4.98E-12	2.04E+06	-3.20E+00	-1.18E-07	1405.72	2452.90	2827.15	0.520	29.82	3660459
132	5.0	14.9	3.17	1.51E-12	6.13E+07	-3.17E+00	-3.57E-08	1374.00	2452.90	2811.51	0.511	29.36	3674516
133	10.0	14.9	3.17	4.66E-13	1.88E+07	-3.15E+00	-1.10E-08	1342.52	2452.90	2796.26	0.501	28.69	3688256
134	10.0	14.9	3.15	1.49E-13	5.94E+08	-3.13E+00	-3.50E-09	1311.26	2452.90	2781.39	0.491	28.13	3701682
135	10.0	14.9	3.13	1.49E-13	1.92E+08	-3.10E+00	-1.14E-09	1280.22	2452.90	2766.89	0.481	27.56	3714794
136	10.0	14.9	3.10	4.89E-14	1.92E+08	-3.08E+00	-3.81E-10	1249.40	2452.90	2752.77	0.471	26.99	3727596
137	10.0	14.9	3.08	1.64E-14	6.41E+09	-3.08E+00	-1.31E-10	1218.78	2452.90	2739.01	0.461	26.42	3740090
138	10.0	14.9	3.06	5.66E-15	2.19E+09	-3.06E+00	-3.71E-10	1203.01	2452.90	2725.61	0.451	25.85	3752278
139	10.0	14.9	3.04	2.01E-15	7.68E+10	-3.04E+00	-4.62E-11	1188.36	2452.90	2721.56	0.441	25.27	3764162
140	10.0	14.9	3.02	7.32E-16	2.77E+10	-3.02E+00	-1.67E-11	1158.13	2452.90	2712.56	0.430	24.70	3775743
141	10.0	14.9	3.00	2.73E-16	1.02E+10	-3.00E+00	-6.21E-12	1128.09	2452.90	2699.87	0.431	24.12	3787024
142	10.0	14.9	2.99	1.05E-16	3.88E+11	-2.99E+00	-2.37E-12	1098.22	2452.90	2687.53	0.421	23.54	3798006
143	10.0	14.9	2.97	4.11E-17	1.51E+11	-2.97E+00	-9.25E-13	1068.53	2452.90	2675.54	0.411	22.96	3808692
144	10.0	14.9	2.95	1.65E-17	6.03E+12	-2.95E+00	-3.71E-13	1039.01	2452.90	2663.88	0.401	22.37	3819082
145	10.0	14.9	2.94	6.83E-18	2.47E+12	-2.94E+00	-1.52E-13	1009.64	2452.90	2652.57	0.390	21.77	3829178
146	10.0	14.9	2.92	2.89E-18	1.04E+12	-2.92E+00	-6.43E-14	980.43	2452.90	2641.59	0.380	21.20	3838982
147	10.0	14.9	2.91	1.26E-18	4.46E+12	-2.91E+00	-2.78E-14	951.37	2452.90	2630.94	0.370	20.61	3848496
148	10.0	14.9	2.89	5.59E-19	1.97E+13	-2.89E+00	-1.23E-14	922.45	2452.90	2620.62	0.360	19.54	3857721
149	10.0	14.9	2.88	2.55E-19	8.91E+14	-2.88E+00	-5.60E-15	893.67	2452.90	2610.63	0.349	19.02	3866657
150	10.0	14.9	2.86	1.19E-19	4.13E+14	-2.86E+00	-2.61E-15	865.03	2452.90	2600.96	0.339	19.43	3866657
151	10.0	14.9	2.85	5.71E-20	1.96E-14	-2.85E+00	-1.24E-15	836.51	2452.90	2591.62	0.329	18.83	3875308
152	10.0	14.9	2.84	2.80E-20	9.57E-15	-2.84E+00	-6.08E-16	808.11	2452.90	2582.59	0.318	18.23	3881673
153	10.0	14.9	2.83	1.41E-20	4.78E-15	-2.83E+00	-3.05E-16	779.83	2452.90	2573.88	0.308	17.64	3891754
154	10.0	14.9	2.82	7.25E-21	2.44E-15	-2.82E+00	-1.56E-16	751.67	2452.90	2565.49	0.297	17.04	3899552
155	10.0	14.9	2.81	3.82E-21	1.28E-15	-2.81E+00	-8.23E-17	723.61	2452.90	2557.41	0.287	16.44	3907069
156	10.0	14.9	2.80	2.07E-21	6.87E-16	-2.80E+00	-4.43E-17	695.65	2452.90	2549.64	0.276	15.83	3914035
157	10.0	14.9	2.79	1.14E-21	3.78E-16	-2.79E+00	-2.44E-17	667.80	2452.90	2542.18	0.266	15.23	3921261
158	10.0	14.9	2.78	6.47E-22	2.13E-16	-2.78E+00	-1.38E-17	640.04	2452.90	2535.03	0.255	14.62	3927939
159	10.0	14.9	2.77	3.75E-22	1.23E-16	-2.77E+00	-7.97E-18	612.37	2452.90	2528.19	0.245	14.02	3934340
160	10.0	14.9	2.76	2.23E-22	7.25E-17</td								

**ASCENT61.XLS Ascent in Actual Atmosphere (Launch Angle 61 Degrees)**

	N	O	P	Q	R	S	T
1							
2							
3							
4							
5							
6							
7							
8							
9							
10	r	Alt	Alt	Theta	Theta	Time	Time
11	(km)	(m)	(km)	(Rad)	(Deg)	(sec)	(min)
12	4002.9	600906.5	600.9	0.1	4.0	647.0	10.8
13							
14							
15							
16							
17							
18							
19							
20	r	Alt	Alt	Theta	Theta	Time	Time
21	(km)	(m)	(km)	(Rad)	(Deg)	(sec)	(min)
22	3397.8	-4200	-4.20	0.00	0.00	0.0	0.00
23	3397.8	-4181	-4.18	0.00	0.00	1.0	0.02
24	3397.9	-4142	-4.14	0.00	0.00	2.0	0.03
25	3397.9	-4084	-4.08	0.00	0.00	3.0	0.05
26	3398.0	-4008	-4.01	0.00	0.00	4.0	0.07
27	3398.1	-3914	-3.91	0.00	0.00	5.0	0.08
28	3398.2	-3801	-3.80	0.00	0.01	6.0	0.10
29	3398.3	-3671	-3.67	0.00	0.01	7.0	0.12
30	3398.5	-3524	-3.52	0.00	0.01	8.0	0.13
31	3398.6	-3359	-3.36	0.00	0.01	9.0	0.15
32	3398.8	-3176	-3.18	0.00	0.02	10.0	0.17
33	3399.0	-2976	-2.98	0.00	0.02	11.0	0.18
34	3399.2	-2760	-2.76	0.00	0.02	12.0	0.20
35	3399.5	-2526	-2.53	0.00	0.03	13.0	0.22
36	3399.7	-2276	-2.28	0.00	0.03	14.0	0.23
37	3400.0	-2010	-2.01	0.00	0.04	15.0	0.25
38	3400.3	-1726	-1.73	0.00	0.04	16.0	0.27
39	3400.6	-1427	-1.43	0.00	0.05	17.0	0.28
40	3400.9	-1112	-1.11	0.00	0.05	18.0	0.30
41	3401.2	-781	-0.78	0.00	0.06	19.0	0.32
42	3401.6	-434	-0.43	0.00	0.06	20.0	0.33
43	3401.9	-71	-0.07	0.00	0.07	21.0	0.35
44	3402.3	307	0.31	0.00	0.08	22.0	0.37
45	3402.7	700	0.70	0.00	0.08	23.0	0.38
46	3403.1	1109	1.11	0.00	0.09	24.0	0.40
47	3403.5	1533	1.53	0.00	0.10	25.0	0.42
48	3404.0	1972	1.97	0.00	0.11	26.0	0.43
49	3404.4	2426	2.43	0.00	0.12	27.0	0.45
50	3404.9	2895	2.90	0.00	0.12	28.0	0.47
51	3405.4	3379	3.38	0.00	0.13	29.0	0.48
52	3405.9	3878	3.88	0.00	0.14	30.0	0.50
53	3406.4	4392	4.39	0.00	0.15	31.0	0.52
54	3406.9	4921	4.92	0.00	0.16	32.0	0.53
55	3407.5	5465	5.46	0.00	0.17	33.0	0.55
56	3408.0	6024	6.02	0.00	0.18	34.0	0.57
57	3408.6	6598	6.60	0.00	0.20	35.0	0.58
58	3409.2	7187	7.19	0.00	0.21	36.0	0.60
59	3409.8	7792	7.79	0.00	0.22	37.0	0.62
60	3410.4	8413	8.41	0.00	0.23	38.0	0.63
61	3411.0	9049	9.05	0.00	0.24	39.0	0.65
62	3411.7	9701	9.70	0.00	0.26	40.0	0.67
63	3412.4	10369	10.37	0.00	0.27	41.0	0.68
64	3413.1	11053	11.05	0.00	0.29	42.0	0.70
65	3413.8	11754	11.75	0.01	0.30	43.0	0.72
66	3414.5	12471	12.47	0.01	0.31	44.0	0.73
67	3415.2	13206	13.21	0.01	0.33	45.0	0.75
68	3416.0	13959	13.96	0.01	0.34	46.0	0.77
69	3416.7	14729	14.73	0.01	0.36	47.0	0.78
70	3417.5	15517	15.52	0.01	0.38	48.0	0.80
71	3418.3	16324	16.32	0.01	0.39	49.0	0.82
72	3419.1	17150	17.15	0.01	0.41	50.0	0.83
73	3420.0	17995	18.00	0.01	0.43	51.0	0.85
74	3420.9	18860	18.86	0.01	0.45	52.0	0.87
75	3421.7	19746	19.75	0.01	0.47	53.0	0.88
76	3422.7	20652	20.65	0.01	0.49	54.0	0.90
77	3423.6	21580	21.58	0.01	0.51	55.0	0.92
78	3424.5	22530	22.53	0.01	0.53	56.0	0.93
79	3425.5	23503	23.50	0.01	0.55	57.0	0.95
80	3426.5	24499	24.50	0.01	0.57	58.0	0.97
81	3427.5	25518	25.52	0.01	0.59	59.0	0.98
82	3428.6	26563	26.56	0.01	0.61	60.0	1.00
83	3429.6	27632	27.63	0.01	0.64	61.0	1.02
84	3430.7	28728	28.73	0.01	0.66	62.0	1.03
85	3431.9	29851	29.85	0.01	0.69	63.0	1.05
86	3433.0	31001	31.00	0.01	0.71	64.0	1.07
87	3434.2	32180	32.18	0.01	0.74	65.0	1.08
88	3435.4	33289	33.39	0.01	0.76	66.0	1.09
89	3436.6	34628	34.63	0.01	0.79	67.0	1.12
90	3437.9	35898	35.90	0.01	0.82	68.0	1.13

## ASCENT61.XLS Ascent in Actual Atmosphere (Launch Angle 61 Degrees)

	N	O	P	Q	R	S	T
91	3439.2	37201	37.20	0.01	0.85	69.0	1.15
92	3440.5	38537	38.54	0.02	0.88	70.0	1.17
93	3441.9	39908	39.91	0.02	0.91	71.0	1.18
94	3443.3	41314	41.31	0.02	0.94	72.0	1.20
95	3444.8	42758	42.76	0.02	0.97	73.0	1.22
96	3446.2	44239	44.24	0.02	1.01	74.0	1.23
97	3447.8	45761	45.76	0.02	1.04	75.0	1.25
98	3449.3	47323	47.32	0.02	1.08	76.0	1.27
99	3450.9	48928	48.93	0.02	1.11	77.0	1.28
100	3452.6	50577	50.58	0.02	1.15	78.0	1.30
101	3454.3	52271	52.27	0.02	1.19	79.0	1.32
102	3456.0	54013	54.01	0.02	1.23	80.0	1.33
103	3457.8	55804	55.80	0.02	1.27	81.0	1.35
104	3459.6	57646	57.65	0.02	1.31	82.0	1.37
105	3461.5	59481	59.48	0.02	1.35	83.0	1.38
106	3463.3	61312	61.31	0.02	1.39	84.0	1.40
107	3465.1	63136	63.14	0.03	1.43	85.0	1.42
108	3467.0	64956	64.96	0.03	1.47	86.0	1.43
109	3476.0	74029	74.03	0.03	1.51	87.0	1.45
110	3485.0	82998	83.00	0.03	1.55	92.0	1.53
111	3493.9	91872	91.87	0.03	1.60	97.0	1.62
112	3502.7	100655	100.65	0.03	1.64	102.0	1.70
113	3511.3	109349	109.35	0.03	1.68	107.0	1.78
114	3520.0	117955	117.96	0.03	1.72	112.0	1.87
115	3528.5	126475	126.47	0.03	1.75	117.0	1.95
116	3536.9	134908	134.91	0.03	1.79	122.0	2.03
117	3545.3	143256	143.26	0.03	1.83	127.0	2.12
118	3561.7	159696	159.70	0.03	1.91	137.0	2.28
119	3569.8	167790	167.79	0.03	1.98	142.0	2.37
120	3577.8	175799	175.80	0.03	1.99	147.0	2.45
121	3585.7	183725	183.72	0.04	2.03	152.0	2.53
122	3593.6	191567	191.57	0.04	2.07	157.0	2.62
123	3601.3	199327	199.33	0.04	2.11	162.0	2.70
124	3609.0	207004	207.00	0.04	2.15	167.0	2.78
125	3616.6	214598	214.60	0.04	2.19	172.0	2.87
126	3624.1	222111	222.11	0.04	2.23	177.0	2.95
127	3631.5	229543	229.54	0.04	2.26	182.0	3.03
128	3638.9	236893	236.89	0.04	2.30	187.0	3.12
129	3646.2	244162	244.16	0.04	2.34	192.0	3.20
130	3653.4	251351	251.35	0.04	2.38	197.0	3.28
131	3660.5	258459	258.46	0.04	2.42	202.0	3.37
132	3674.5	272516	272.52	0.04	2.46	207.0	3.45
133	3688.3	286256	286.26	0.04	2.49	217.0	3.62
134	3701.7	299682	299.68	0.04	2.53	227.0	3.78
135	3714.8	312794	312.79	0.04	2.57	237.0	3.95
136	3727.6	325596	325.60	0.05	2.61	247.0	4.12
137	3740.1	338090	338.09	0.05	2.65	257.0	4.28
138	3752.3	350278	350.28	0.05	2.68	267.0	4.45
139	3764.4	362162	362.16	0.05	2.72	277.0	4.62
140	3775.7	373743	373.74	0.05	2.76	287.0	4.78
141	3787.0	385024	385.02	0.05	2.80	297.0	4.95
142	3798.0	396006	396.01	0.05	2.83	307.0	5.12
143	3808.7	406692	406.69	0.05	2.87	317.0	5.28
144	3819.1	417082	417.08	0.05	2.91	327.0	5.45
145	3829.2	427178	427.18	0.05	2.94	337.0	5.62
146	3839.0	436982	436.98	0.05	2.98	347.0	5.78
147	3848.5	446496	446.50	0.05	3.02	357.0	5.95
148	3857.7	455721	455.72	0.05	3.05	367.0	6.12
149	3866.7	464657	464.66	0.05	3.09	377.0	6.28
150	3875.3	473308	473.31	0.05	3.12	387.0	6.45
151	3883.7	481673	481.67	0.06	3.16	397.0	6.62
152	3891.8	489754	489.75	0.06	3.20	407.0	6.78
153	3899.6	497552	497.55	0.06	3.23	417.0	6.95
154	3907.1	505069	505.07	0.06	3.27	427.0	7.12
155	3914.3	512305	512.30	0.06	3.30	437.0	7.28
156	3921.3	519261	519.26	0.06	3.34	447.0	7.45
157	3927.9	525939	525.94	0.06	3.38	457.0	7.62
158	3934.3	532340	532.34	0.06	3.41	467.0	7.78
159	3940.5	538463	538.46	0.06	3.45	477.0	7.95
160	3946.3	544311	544.31	0.06	3.48	487.0	8.12
161	3951.9	549884	549.88	0.06	3.52	497.0	8.28
162	3957.2	555183	555.18	0.06	3.55	507.0	8.45
163	3962.2	560208	560.21	0.06	3.59	517.0	8.62
164	3967.0	564960	564.96	0.06	3.63	527.0	8.78
165	3971.4	569440	569.44	0.06	3.66	537.0	8.95
166	3975.6	573649	573.65	0.06	3.70	547.0	9.12
167	3979.6	577586	577.59	0.07	3.73	557.0	9.28
168	3983.3	581253	581.25	0.07	3.77	567.0	9.45
169	3986.7	584651	584.65	0.07	3.80	577.0	9.62
170	3989.8	587778	587.78	0.07	3.84	587.0	9.78
171	3992.6	590637	590.64	0.07	3.87	597.0	9.95
172	3995.2	593227	593.23	0.07	3.91	607.0	10.12
173	3997.5	595549	595.55	0.07	3.94	617.0	10.28
174	4001.4	599388	599.39	0.07	4.01	637.0	10.62
175	4002.9	600907	600.91	0.07	4.05	647.0	10.78
176	4004.2	602158	602.16	0.07	4.08	657.0	10.95
177	4005.1	603142	603.14	0.07	4.12	667.0	11.12

ASC\_DELV.XLS

DELTA V'S DURING ASCENT BURN

0.508 Mass Depetion Rate (kg/sec)			
1500.0 Thrust (N)			
time (sec)	mass (kg)	Acel (m/s^2)	Delta V (m/s)
0	56.800	26.408	26.408
1	56.292	26.647	53.055
2	55.784	26.889	79.945
3	55.276	27.137	107.081
4	54.768	27.388	134.470
5	54.260	27.645	162.115
6	53.752	27.906	190.021
7	53.244	28.172	218.193
8	52.735	28.444	246.637
9	52.227	28.721	275.358
10	51.719	29.003	304.360
11	51.211	29.290	333.651
12	50.703	29.584	363.235
13	50.195	29.883	393.118
14	49.687	30.189	423.307
15	49.179	30.501	453.808
16	48.671	30.819	484.627
17	48.163	31.144	515.771
18	47.655	31.476	547.248
19	47.147	31.816	579.063
20	46.639	32.162	611.225
21	46.131	32.516	643.742
22	45.623	32.878	676.620
23	45.115	33.249	709.869
24	44.606	33.627	743.496
25	44.098	34.015	777.511
26	43.590	34.411	811.922
27	43.082	34.817	846.739
28	42.574	35.233	881.972
29	42.066	35.658	917.630
30	41.558	36.094	953.724
31	41.050	36.541	990.265
32	40.542	36.999	1027.264
33	40.034	37.468	1064.732
34	39.526	37.950	1102.682
35	39.018	38.444	1141.126
36	38.510	38.951	1180.077
37	38.002	39.472	1219.549
38	37.494	40.007	1259.556
39	36.985	40.556	1300.113
40	36.477	41.121	1341.234
41	35.969	41.702	1382.936

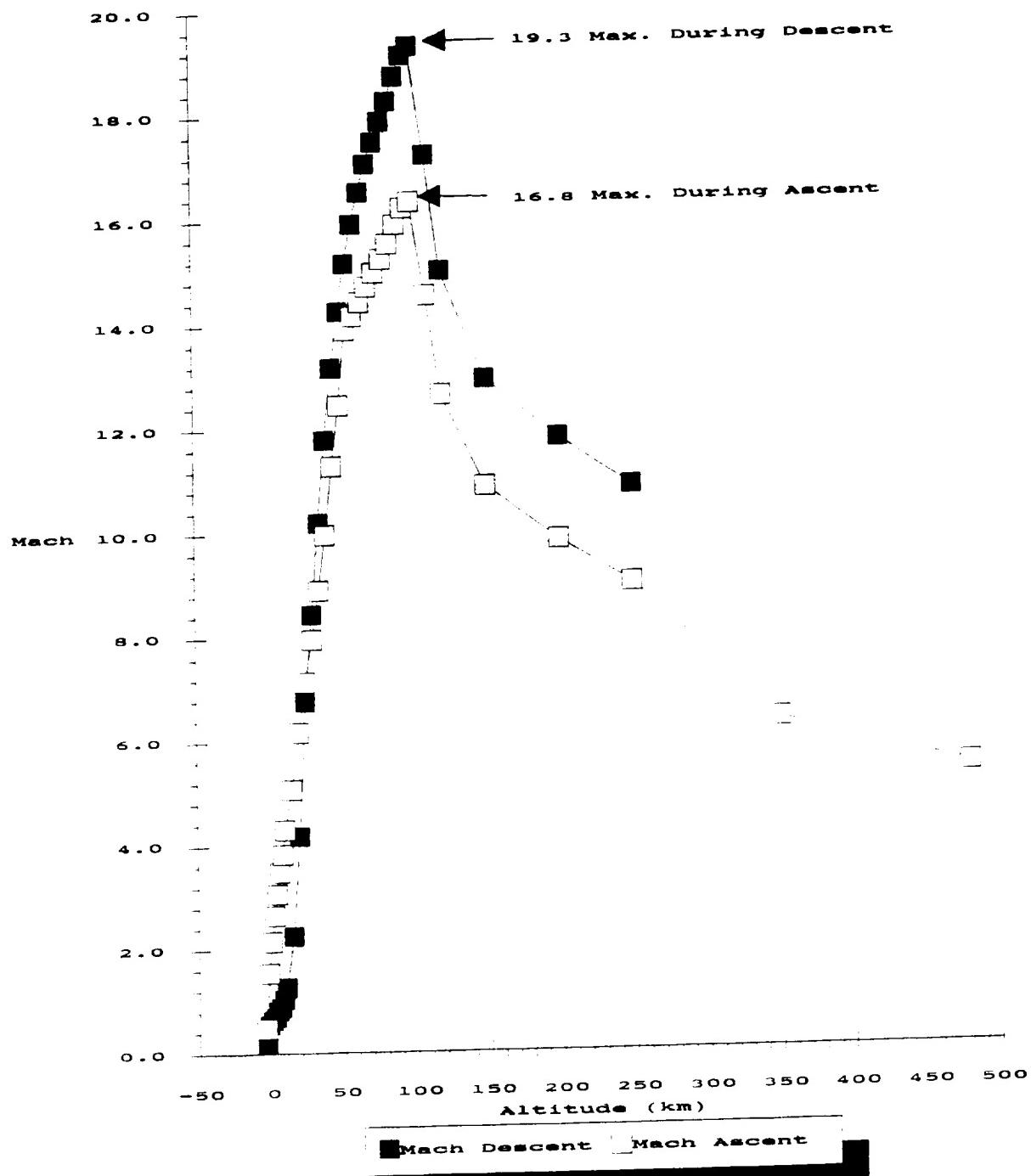
time (sec)	mass (kg)	Acel (m/s^2)	Delta V (m/s)
42	35.461	42.300	1425.236
43	34.953	42.914	1468.150
44	34.445	43.547	1511.698
45	33.937	44.199	1555.897
46	33.429	44.871	1600.768
47	32.921	45.564	1646.332
48	32.413	46.278	1692.610
49	31.905	47.015	1739.625
50	31.397	47.776	1787.400
51	30.889	48.561	1835.962
52	30.381	49.374	1885.335
53	29.873	50.213	1935.548
54	29.365	51.082	1986.631
55	28.856	51.981	2038.612
56	28.348	52.913	2091.525
57	27.840	53.879	2145.404
58	27.332	54.880	2200.284
59	26.824	55.920	2256.204
60	26.316	56.999	2313.203
61	25.808	58.121	2371.324
62	25.300	59.289	2430.613
63	24.792	60.504	2491.116
64	24.284	61.769	2552.886
65	23.776	63.089	2615.975
66	23.268	64.467	2680.442
67	22.760	65.906	2746.348
68	22.252	67.411	2813.759
69	21.744	68.986	2882.745
70	21.235	70.636	2953.381
71	20.727	72.368	3025.749
72	20.219	74.186	3099.936
73	19.711	76.099	3176.034
74	19.203	78.112	3254.146
75	18.695	80.235	3334.381
76	18.187	82.476	3416.857
77	17.679	84.846	3501.703
78	17.171	87.357	3589.060
79	16.663	90.020	3679.080
80	16.155	92.851	3771.931
81	15.647	95.866	3867.798
82	15.139	99.084	3966.882
83	14.631	102.525	4069.406

Interpolate to find time at end of burn 0.269

Use average of 82 and 83 second accelerations 0.508 100.804 53.442

0.530	4020.324
Delta V total at Wt.=14.9 (in m/s)	

Mach Number vs. Altitude



MACH\_&\_Q.XLS Descent Mach Numbers and Dynamic Pressures

Alt (km)	Density (kg/m^3)	Speed of Sound(m/s)	Descent			Ascent		
			v (m/s)	Mach	q inf (N/m^2)	v (m/s)	Mach	q inf (N/m^2)
-4	1.35E-02	255	44	0.2	13	119	0.5	96
-2	1.18E-02	253	140	0.6	116	191	1.5	902
0	1.02E-02	252	160	0.6	131	540	2.1	1487
2	8.89E-03	251	174	0.7	135	663	2.6	1954
4	7.72E-03	249	203	0.8	159	763	3.1	2247
6	6.67E-03	248	217	0.9	157	865	3.5	2495
8	5.74E-03	247	254	1.0	185	944	3.8	2558
10	4.91E-03	246	311	1.3	237	1055	4.3	2732
15	3.34E-03	244	548	2.2	502	1236	5.1	2551
20	2.28E-03	240	1002	4.2	1145	1481	6.2	2500
25	1.53E-03	236	1596	6.8	1949	1686	7.1	2175
30	9.99E-04	235	1985	8.4	1968	1872	8.0	1750
35	6.49E-04	234	2387	10.2	1849	2083	8.9	1408
40	4.19E-04	233	2747	11.8	1581	2322	10.0	1130
45	2.74E-04	230	3031	13.2	1259	2595	11.3	923
50	1.76E-04	227	3234	14.2	920	2826	12.4	703
55	1.13E-04	222	3368	15.2	641	3084	13.9	537
60	7.12E-05	217	3453	15.9	424	3072	14.2	336
65	4.43E-05	212	3505	16.5	272	3058	14.4	207
70	2.72E-05	207	3535	17.1	170	3048	14.7	126
75	1.64E-05	203	3551	17.5	103	3402	16.8	95
80	9.75E-06	199	3560	17.9	62	3035	15.3	45
85	5.74E-06	195	3563	18.3	36	3030	15.5	26
90	3.32E-06	190	3563	18.8	21	3022	15.9	15
95	1.88E-06	186	3561	19.1	12	3018	16.2	9
100	1.02E-06	184	3555	19.3	6	3007	16.3	5
110	2.75E-07	206	3550	17.2	2	2997	14.5	1
120	8.72E-08	236	3545	15.0	1	2985	12.6	0
130	3.52E-08	252						
140	1.61E-08	264						
150	8.11E-09	272	3518	12.9	0	2958	10.9	0
160	4.34E-09	277						
170	2.41E-09	281						
180	1.34E-09	288						
190	7.86E-10	291						
200	4.66E-10	294	3473	11.8	0	2894	9.8	0
210	2.81E-10	297						
220	1.71E-10	301						
230	1.06E-10	305						
240	6.60E-11	310						
250	4.17E-11	315	3421	10.9	0	2835	9.0	0
260	2.66E-11	324						
270	1.73E-11	332						
280	1.14E-11	343						
290	7.63E-12	354						
300	5.22E-12	367						
310	3.65E-12	380						
320	2.61E-12	394						
330	1.91E-12	407						
340	1.43E-12	419						
350	1.09E-12	430				2726	6.3	0
480	9.03E-14	480				2583	5.4	0

Section E:  
**Specialized Structures**

Tim Gibson

## Parachute

During the descent through the martian atmosphere, the lander will employ a parachute to deaccelerate to a velocity at which the liquid engine can prepare the lander for touchdown. The size of the parachute, measured by its reference area, the area of a circle whose diameter is equal to the distance across the fully deployed parachute, must be large enough to create enough drag so that the required final or "terminal" velocity will be attained, without too much deacceleration during the opening "jerk". It can be estimated from Air Force research (ref. 3) that the peak deacceleration during opening jerk is slightly more than 3 times that of the deacceleration calculated from a force-balance relationship. The lander is limited to a maximum acceleration of 10 earth g's. This limits maximum force-balance calculated deacceleration to under 3.33 g's.

The force-balance relationship is:

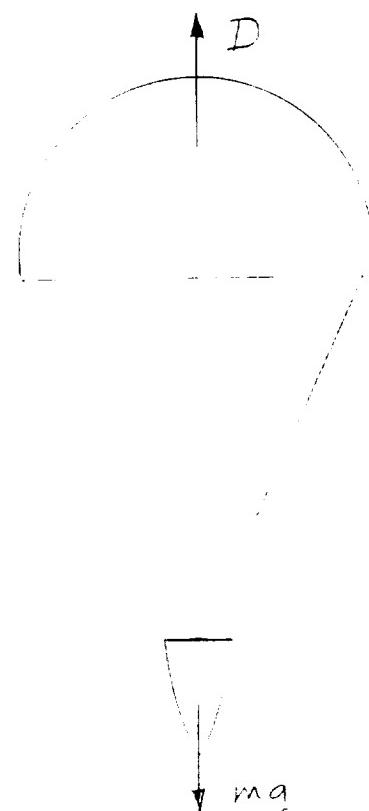


DIAGRAM DRAWN IN  
OF POOR QUALITY

which yields:  $D - mg = ma$

where: drag,  $D = C_D \frac{1}{2} \rho v^2 S$

drag coefficient,  $C_D$

atmosphere density,  $\rho$

velocity,  $V$

reference area,  $S$

mass of the lander,  $m$

deacceleration,  $a$

so, we have:  $C_D \frac{1}{2} \rho v^2 S - mg = ma$

From this, by setting  $a = 0$ , we can obtain the reference area required for a desired terminal velocity:

$$S = \frac{2mg}{C_D \rho V^2}$$

or, the maximum force-balance deacceleration:

$$a = \frac{(CD \frac{1}{2} \rho v^2 S - mg)}{m}$$

where  $V$  is the deployment velocity.

By starting at  $V$  = deployment velocity, and recalculating  $V$  every time interval  $\Delta t$  by numerically integrating the acceleration (see Appendix E.A.), the maximum descent distance and time to reach terminal velocity can be determined. This method yields a maximum because the actual acceleration rate is, as mentioned earlier, higher than the force-balance method predicts. However, the maximum is desirable for determining the minimum altitude at which the parachute can be deployed safely. Low altitude parachute deployment is desirable to minimize the effect of cross winds.

Initially, the lander was to use a 2-stage parachute. The small first stage, 3.33 m diameter, limited peak deacceleration to 3.14 g's at a deployment velocity of 280.00 m/sec at 1 km altitude (fig. E / ). The larger second stage, 9.9 m in diameter, limited peak deacceleration to 2.69 g's at a deployment velocity of 88.00 m/sec (fig. E / ) and attained a terminal velocity of less than 30.00 m/sec

in less than 75 m. This configuration could be deployed at less than 1 km above ground level and maintain a 3 times maximum g's value under our 10g mission limit.

The parachute is an extended skirt canopy type (fig E3) which has a higher drag than flat circular or cross-form types but has a longer opening time and correspondingly has lower peak opening forces. The canopy is fabricated of 2.60 ounce per square yard ( $89 \text{ g/m}^2$ ) dacron cloth, per MIL-C-7350, type 1. The suspension lines are tubular braided dacron with an ultimate tensile strength of 2740N. The approximate weight of the canopy can be calculated from the surface area of 1/2 of a sphere:

$$A_{\text{surface}} = 2 \pi r^2$$

and a required storage volume (chute pack) based on a maximum packing density of  $\approx 600 \text{ kg/m}^3$ .

So, the mass of the chute canopy,  $m_{\text{canopy}} = 89 A_{\text{surface}} = 89 (2 \pi r^2)$ ,

and the volume of the chute pack,  $V_{\text{pack}} = (89) (2 \pi r^2) / 600 \times 10^3$ .

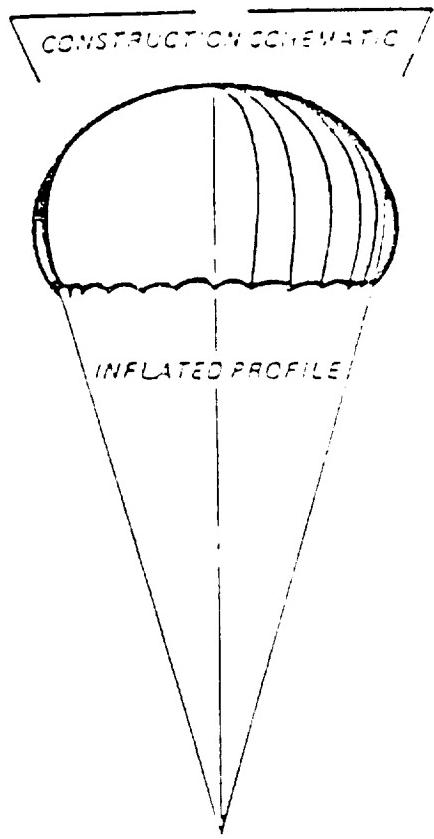
This mass and volume dependence on the radius squared led to an optimization where the weight and size of the parachute were compared to the weight of added liquid fuel to accomplish the same deacceleration. It was determined that decreasing the parachute from 9.9 meters in diameter ( $m_{\text{canopy}} = 13.7 \text{ kg}$ ) to 4 meters in diameter ( $m_{\text{canopy}} = 0.75 \text{ kg}$ ), while adding  $\approx 4 \text{ kg}$  of fuel, would decrease mass by  $\approx 8.9 \text{ kg}$ . Also, the volume required for the chute pack went from  $0.0228 \text{ m}^3$  to  $0.00125 \text{ m}^3$ .

Using the 4 m diameter parachute, an analysis was done (fig. E4) for the earliest possible deployment, 5 km above ground level, with the highest deployment velocity possible,  $V = 248.54 \text{ m/sec}$ . Maximum force-balance deacceleration was 2.47 g's, with a distance to terminal velocity of  $\approx 0.5 \text{ km}$ . Next, an analysis was done (fig. E5) for a late deployment, -3 km altitude, with a corresponding terminal velocity of 187.4 m/sec. Maximum force-

balance deacceleration was 2.48 g's, with a distance to terminal velocity of  $\approx$  325 m.

FIG. E3

Description  
EXTENDED SKIRT CANOPY



© 1987  
Ballistic Research Laboratory

## APPENDIX EA

```
program mars
implicit none
real*4 cd, rho, s, g, m, v, vterm, t, h, a, vnew, accel
real*4 deltat
integer*4 i

c
cd = 1.0
deltat = 0.1
rho = 0.0127
s = 12.5664
g = 3.73
m = 100.0
v = 187.4
vterm = sqrt((2.0*m*g)/(cd*rho*s))

c
50 write(6,50) vterm
   format(1x,'v term = ', f7.3,'m/sec',/,1x,'V (m/sec)',
*      5x,'h (meters)',5x,'accel (g s)',
*      5x,'time (sec)')

c
t = 0.0
h = 0.0
do 100 i = 1, 200
   t = t + deltat
   a = (0.5*cd*rho*(v**2)*s - m*g)/m
   h = h + v*deltat - 0.5*a*(deltat**2)
   vnew = sqrt((v**2) - 2.0*a*h)
   accel = a/9.81
   write(6,60) v, h, accel, t
60   format(1x,f7.3,7x,f7.2,10x,f7.5,8x,f5.1)
   if (v .le. (1.00001*vterm)) go to 200
   v = vnew
100 enddo
200 continue
end
```

SHOOTED DUCKS IS  
OF POOR QUALITY

FIG. E1

V (m/sec)	h (meters)	accel (g s)	time (sec)
280.000	27.85	3.14891	.1
276.911	55.39	3.07227	.2
270.816	82.32	2.92355	.3
261.952	108.39	2.71320	.4
250.698	133.34	2.45618	.5
237.537	156.98	2.16995	.6
223.025	179.19	1.87222	.7
207.745	199.89	1.57901	.8
192.264	219.05	1.30315	.9
177.101	236.71	1.05366	1.0
162.700	252.94	.83567	1.1
149.412	267.85	.65093	1.2
137.489	281.57	.49855	1.3
127.079	294.26	.37586	1.4
118.233	306.07	.27919	1.5
110.917	317.15	.20451	1.6
105.024	327.65	.14782	1.7
100.398	337.68	.10549	1.8
96.855	347.37	.07435	1.9
94.202	356.78	.05178	2.0
92.258	366.01	.03563	2.1
90.861	375.09	.02423	2.2
89.874	384.08	.01629	2.3
89.189	393.00	.01082	2.4
88.720	401.87	.00710	2.5
88.403	410.71	.00461	2.6
88.193	419.53	.00295	2.7
88.055	428.33	.00187	2.8
87.966	437.13	.00117	2.9
87.909	445.92	.00072	3.0
87.873	454.71	.00044	3.1
87.850	463.49	.00027	3.2
87.836	472.28	.00016	3.3
87.828	481.06	.00009	3.4
87.823	489.84	.00005	3.5
87.820	498.62	.00003	3.6
87.818	507.41	.00002	3.7
87.817	516.19	.00001	3.8
87.817	524.97	.00001	3.9

FIG. E 2

v (m/sec)	h (meters)	accel (g s)	time (sec)
88.000	8.67	2.69566	.1
85.356	17.08	2.51574	.2
80.265	25.00	2.18488	.3
73.286	32.24	1.76431	.4
65.229	38.70	1.32630	.5
56.989	44.35	.93109	.6
49.372	49.26	.61311	.7
42.953	53.54	.38056	.8
38.017	57.33	.22368	.9
34.550	60.78	.12494	1.0
32.322	64.00	.06647	1.1
31.004	67.10	.03371	1.2
30.279	70.13	.01629	1.3
29.907	73.12	.00749	1.4
29.727	76.09	.00328	1.5
29.644	79.06	.00136	1.6
29.609	82.02	.00053	1.7
29.594	84.98	.00020	1.8
29.589	87.94	.00007	1.9
29.587	90.90	.00002	2.0
29.586	93.85	.00001	2.1

FIG. E4

v term = 90.802m/sec	v (m/sec)	h (meters)	accel (g s)	time (sec)
	248.540	24.73	2.46841	.1
	246.118	49.23	2.41317	.2
	241.337	73.25	2.30569	.3
	234.372	96.58	2.15289	.4
	225.501	119.03	1.96476	.5
	215.086	140.45	1.75316	.6
	203.546	160.73	1.53037	.7
	191.323	179.80	1.30781	.8
	178.860	197.63	1.09505	.9
	166.568	214.25	.89924	1.0
	154.806	229.69	.72492	1.1
	143.868	244.05	.57427	1.2
	133.971	257.43	.44746	1.3
	125.253	269.93	.34324	1.4
	117.773	281.70	.25941	1.5
	111.520	292.84	.19329	1.6
	106.424	303.48	.14208	1.7
	102.372	313.71	.10307	1.8
	99.226	323.63	.07381	1.9
	96.835	333.31	.05220	2.0
	95.056	342.81	.03646	2.1
	93.757	352.19	.02515	2.2
	92.826	361.47	.01714	2.3
	92.169	370.69	.01153	2.4
	91.713	379.86	.00766	2.5
	91.401	389.00	.00503	2.6
	91.191	398.12	.00326	2.7
	91.051	407.22	.00209	2.8
	90.960	416.32	.00132	2.9
	90.900	425.41	.00082	3.0
	90.863	434.49	.00050	3.1
	90.839	443.58	.00031	3.2
	90.824	452.66	.00018	3.3
	90.815	461.74	.00011	3.4
	90.810	470.82	.00006	3.5
	90.807	479.90	.00004	3.6
	90.805	488.98	.00002	3.7
	90.804	498.06	.00001	3.8
	90.803	507.14	.00001	3.9

FIG. E 5

v term = 68.369m/sec

v (m/sec)	h (meters)	accel (g s)	time (sec)
187.400	18.62	2.47641	.1
184.971	37.00	2.40283	.2
180.194	54.91	2.26095	.3
173.304	72.14	2.06283	.4
164.666	88.51	1.82535	.5
154.741	103.91	1.56750	.6
144.046	118.25	1.30756	.7
133.100	131.51	1.06080	.8
122.387	143.71	.83816	.9
112.318	154.91	.64593	1.0
103.209	165.20	.48624	1.1
95.269	174.71	.35805	1.2
88.593	183.56	.25821	1.3
83.180	191.87	.18257	1.4
78.940	199.76	.12667	1.5
75.731	207.32	.08628	1.6
73.377	214.66	.05774	1.7
71.701	221.83	.03796	1.8
70.539	228.88	.02452	1.9
69.755	235.85	.01556	2.0
69.236	242.78	.00970	2.1
68.902	249.67	.00594	2.2
68.690	256.54	.00358	2.3
68.559	263.39	.00211	2.4
68.479	270.24	.00122	2.5
68.432	277.08	.00070	2.6
68.404	283.92	.00039	2.7
68.389	290.76	.00021	2.8
68.380	297.60	.00011	2.9
68.375	304.44	.00006	3.0
68.372	311.28	.00003	3.1
68.371	318.11	.00002	3.2
68.370	324.95	.00001	3.3

## Heat Shield Structure

The structure of the heat shield serves to hold the insulation material in position, and serve as an aero-braking device. The 1.5 meter diameter heat shield is designed to withstand a maximum total aero-force of 6.9 kN which occurs at an altitude of 31 km and a corresponding free-stream velocity of 1985 m/sec. The structure in contact with the insulation encounters a maximum temperature of 700 F for approximately 30 seconds.

Due to its high stiffness-to-weight ratio and resistance to thermal effects, a beryllium honeycomb bonded with high-temperature thermoplastic adhesive was chosen for the heat shield structure. Four beryllium tubular braces serve to brace the structure (see fig. E6 ).

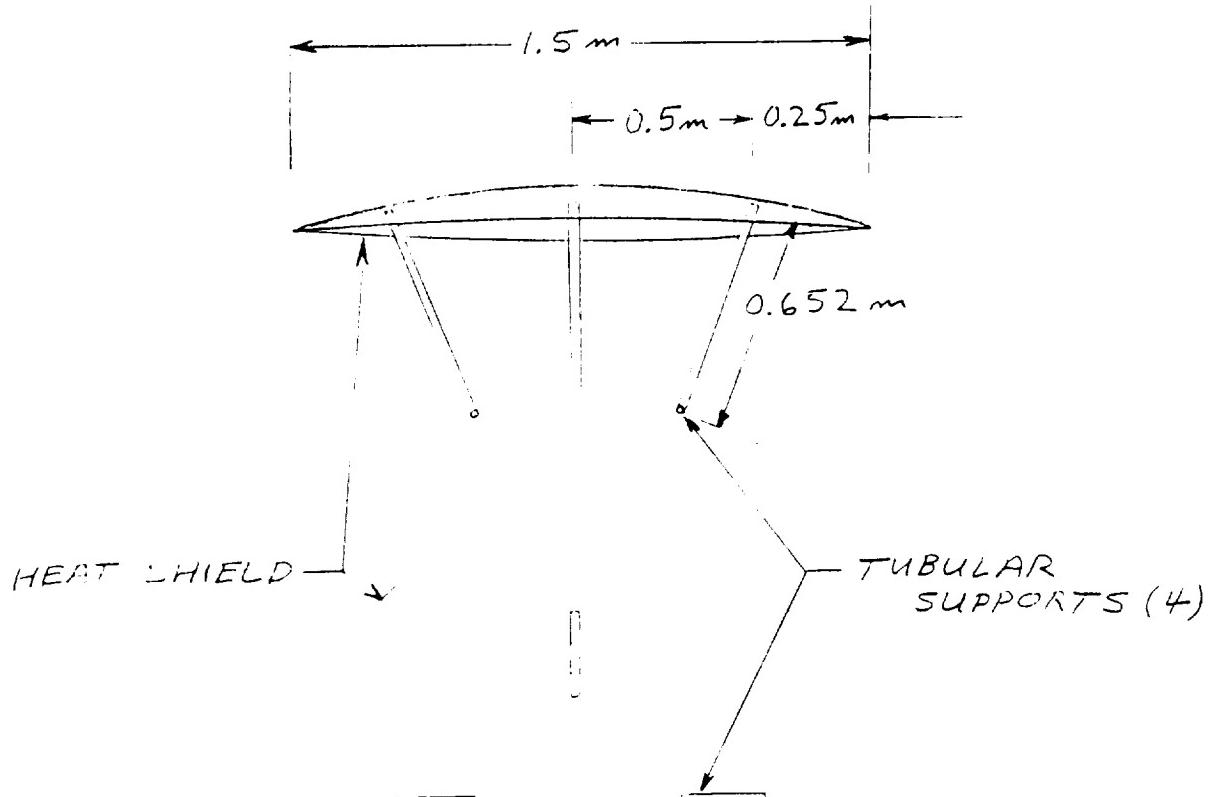
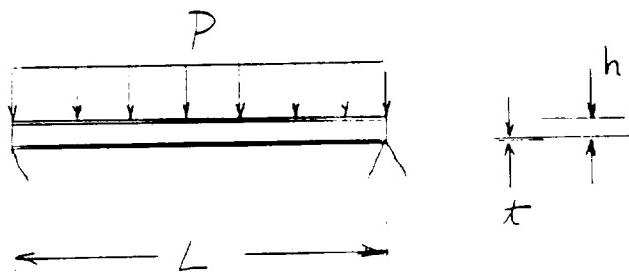


FIG. E6

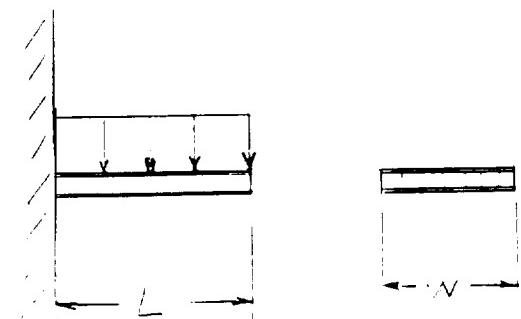
In order to conservatively size the structural components simplified models (see fig. E7) were chosen which would provide an adequate safety margin and facilitate analysis.

FIG. E7

CASE 1



CASE 2



where:  
 unsupported length, L  
 load per unit length, P  
 depth of honeycomb, h  
 thickness of face sheet, t  
 width of beam, w

From strength of materials the stress is: stress =  $M y / I$

$$\text{now, } y = \frac{1}{2}d + t, \quad \text{but since } t \ll \frac{1}{2}h, \quad y = \frac{1}{2}d$$

$$\text{and, by the parallel axis theorem } I = \frac{2}{3}wt^3 + 2wt(\frac{1}{2}d)^2$$

$$\text{since } \frac{2}{3}wt^3 \ll 2wt(\frac{1}{2}d)^2, \text{ we have } I \approx 2wt(\frac{1}{2}d)^2 = \frac{1}{2}wtd^2$$

$$\text{so, stress} \approx M^{(1/2)}h / (\frac{1}{2})wth^2 = M/wtd$$

or, letting  $M$  = moment per unit width, stress =  $M/t_d$

Also from strength of materials we know the maximum moment,  $M$ , of our model beams (fig. E7) for case 1 is equal to

$$M_{(\text{max.})} = PL^2/8 ,$$

and for case 2

$$M_{(\text{max})} = PL^2/8 .$$

Using a maximum unsupported length of  $L = 1 \text{ m}$  for case 1 and  $L = 0.5 \text{ m}$  for case 2, we have:

$$M_{(\text{max})} = P/8$$

Now  $\text{stress}_{\text{crit}}$  for beryllium =  $\text{stress}_{\text{yield}} = 3.24 \times 10^8 \text{ N/m}$  and the distributed load,

$$P = \text{Aero-force}/\text{area} = (6.9 \text{ Kn}/\pi(0.75\text{m})^2)(1\text{m}) = 3905 \text{ N/m}$$

$$\text{Rearranging, we have: } t d = P/8(\text{stress}_y) = 3905/8(3.244 \times 10^8)$$

$$= 1.507 \times 10^{-6}$$

Taking the minimum thickness for the beryllium face sheets to be  $t = 0.010" = 0.000254 \text{ m}$ ,  $\Rightarrow d = 1.507 \times 10^{-6} / 0.000254$   
 $= 0.00593 \text{ m} = 0.23 \text{ in}$

so, we will specify a beryllium honeycomb with a core depth,  $d = 0.25" = 0.00635 \text{ m}$ . Checking, we have

$$\text{stress}_{\text{crit}} = \frac{3905}{8(.00635)(.000254)} = 3.03 \times 10^8 \text{ N/m}^2$$

which is over 6% under stress<sub>y</sub>.

Now, in order to determine the mass of the structure the density of the honeycomb core is needed. A "rule of thumb" for estimating honeycomb

core density is  $\rho_{\text{core}} = 2\rho_{\text{Be}} V_{\text{max}} / \text{stress}_{\text{Be shear}}$

where: density of Be,  $\rho_{\text{Be}} = 1826 \text{ kg/m}^3$

shear stress of Be,  $\text{stress}_s \approx 0.6 \text{ stress}_y$

and the maximum shear stress, again from strength of materials, is

$$V_{\text{max}} = 1/2P = 1.952.5 \text{ N}$$

$$\text{so, } P_{\text{core}} = 2(1826)(1952.5) / 0.6(3.24 \times 10^8) = 0.0376 \text{ kg/m}^3$$

The lightest core material available is approximately 1.5 lbs/ft or 24 kg/m. Apparently the shear force is too low in this application for the "rule of thumb" approximation to be accurate. It does indicate though that it is safe to use the lightweight core. So, the mass of the structure is:

$$\begin{aligned} \text{Mass} &= 2\rho_{\text{Be}}(\pi r^2 t) + \rho_{\text{core}}(\pi r^2 d) \\ &= \pi r^2(2\rho_{\text{Be}}t + \rho_{\text{core}}d) \\ &= \pi(.75)^2[2(1826)(.000254) + 24(.00635)] = 1.91 \text{ kg} \end{aligned}$$

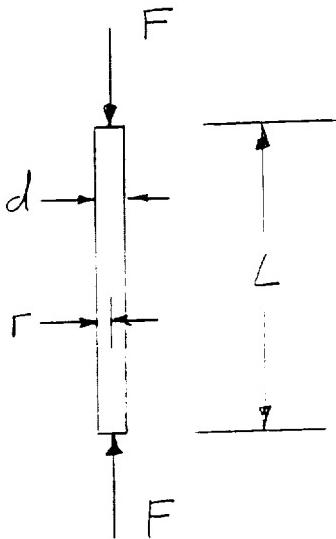
The model for the tubular braces is simple compression (see fig E 3). The force in each tube may be as high as

$$F = (1.5)(\text{max. aero-force} / \# \text{ of tubes})$$

where: max. aero-force = 6.9 kN  
# of tubes = 4

so  $F = (1.5)(6900)/4 = 2588 \text{ N}$

FIG. E8



where:  
diameter,  $d$   
radius,  $r$   
length,  $L$   
wall thickness,  $t$   
force,  $F$

For a thin wall tube under compression there are three design constraints:

1) stress limit: stress =  $F/\pi dt$

2) Euler buckling:  $F = \pi^2 EI/L^2$

3) crippling: stress =  $0.3E^t/r$

For a first iteration, the stress limit and Euler buckling are solved together in order to find the optimum tube dimensions, where

$$I = \pi r^3 t \quad \text{for a thin-wall tube.}$$

Now, solving 1) and 2) for the force,  $F$ , yields

$$(\text{stress})2\pi rt = \frac{\pi^2 E \pi r^3 t}{L^2}$$

$$\text{so } r = \frac{2(\text{stress})L^2}{\pi^2 E} = \frac{2(652)^2(3.24 \times 10^8)}{\pi^2(2.896 \times 10^{11})} \\ = 0.009817 \text{ m.}$$

This is a tube with a diameter of 0.0196 m or 0.773". Solving for the wall thickness,  $t = 0.0001313 \text{ m} = 0.0052 \text{ in.}$  In order to specify a tube which might be commonly available, we will use 0.75 in. (0.01875 m) diameter and 0.010 in. (0.000254 m) wall thickness beryllium tubing. Checking, we have:

$$1) \text{ stress limit, } F = (\text{stress})2\pi rt = 4848 \text{ N}$$

$$2) \text{ Euler buckling, } F = \frac{\pi^2 EI}{L^2} = 4421 \text{ N}$$

$$3) \text{ crippling, stress} = \frac{0.3Et}{r} = 23.5 \times 10^8$$

which are all considerably over any anticipated forces. Now, the mass of

each tube is, mass/tube =  $\pi dt L \rho_{Be}$

$$= \pi(0.01875)(0.000254)(0.652)(1826) = .0178 \text{ kg}$$

So, the total for four tubes is  $4(0.0178) = .0712 \text{ kg}$

Therefore, the mass of the entire structure is:  $1.91 + .0712 = 1.98$   
kg

## Soil Sample Collection

The mission requirement is to bring 1 kg of Martian soil back to the orbiting MOV (Mars Orbiting Vehicle). This will be accomplished with the use of an extendable surface sampler, which consists of a collector head attached to the end of a retractable boom. The boom is constructed from two ribbons of beryllium welded together along the edges. When extended, the two layers opened to form a rigid tube. When retracted, the boom flattens onto a storage spool. A flat cable sandwiched between the boom layers transmits electrical power to the collector head.

The collector head is a scoop with a movable lid. To fill the scoop, the boom is extended along or into the surface. Once full, the lid closes. The boom is then elevated such that the scoop can be dumped into a container at the top of the lander (fig. E9). The nose cap at the top of the lander's aero-shell (fig. E10) is moved upward by a threaded driveshaft and, when free of the aero-shell, is then rotated out the way for dumping into the sample container.

Design considerations include tube buckling and bending stresses, and torque for pushing the scoop into the Martian soil, lifting the sample into the container, and opening and closing the container.

### Data, Martian soil:

density,  $\rho = 0.7 \text{ to } 3.2 \text{ g/cm}^3$ ,

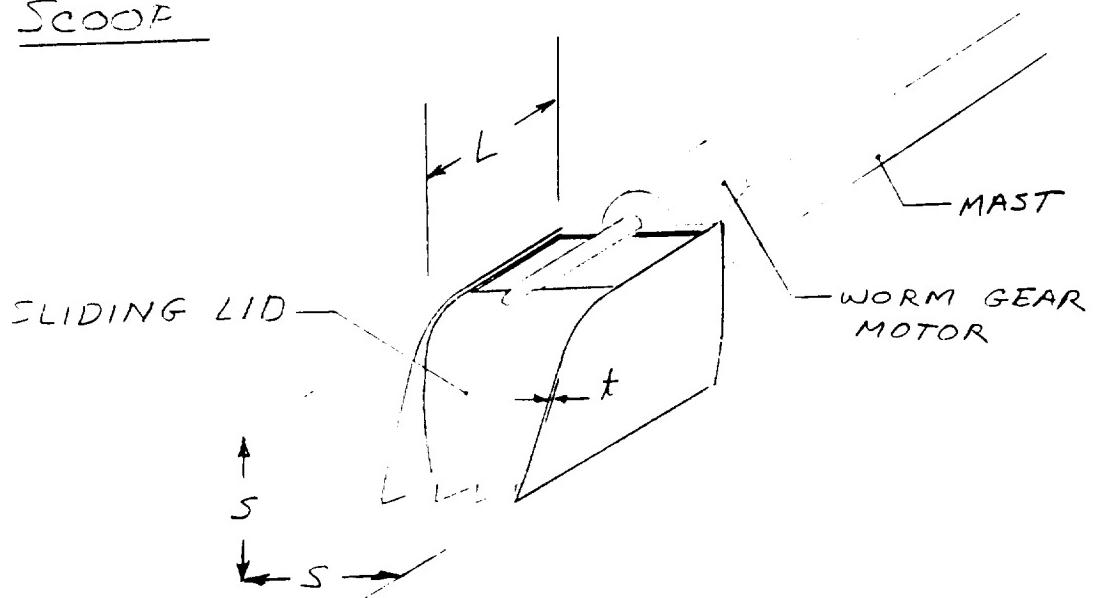
maximum penetration resistance,  $PR = 6 \text{ N/cm}^2 \text{ cm}$

### Scoop

Volume required for 1 kg sample,  $\rho = 0.7 \text{ g/cm}^3$

$$\frac{1}{700} = 0.00143 \text{ m}^3 = 1430 \text{ cm}^3$$

## Scoop

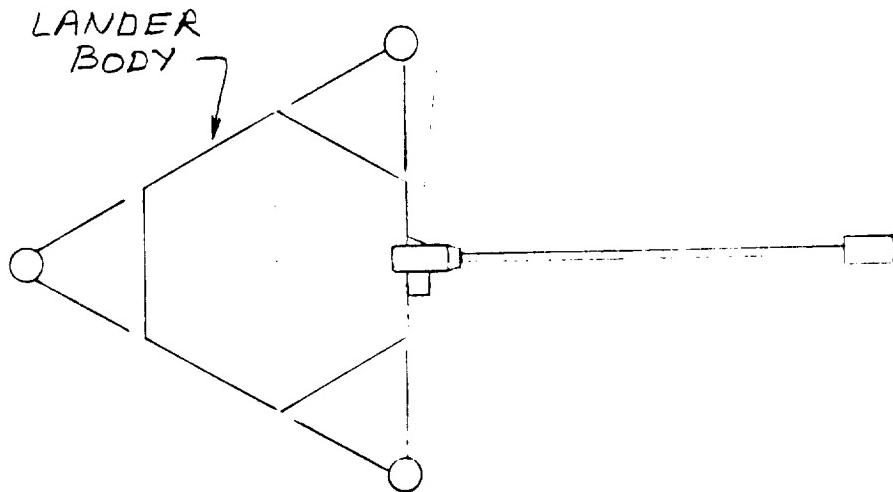


where:  $s$  = scoop side,  $L$  = scoop length,  $t$  = material thickness.

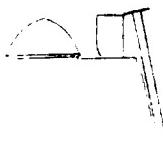
Since the collector is being driven into the soil, titanium is used as a material for the scoop due to its toughness. The worm-gear motor should have enough torque to close the lid against maximum penetration resistance. Now, the force needed to drive the scoop into the soil,

FIG. E9

SAMPLER-ARM OPERATION



DUMP POSITION

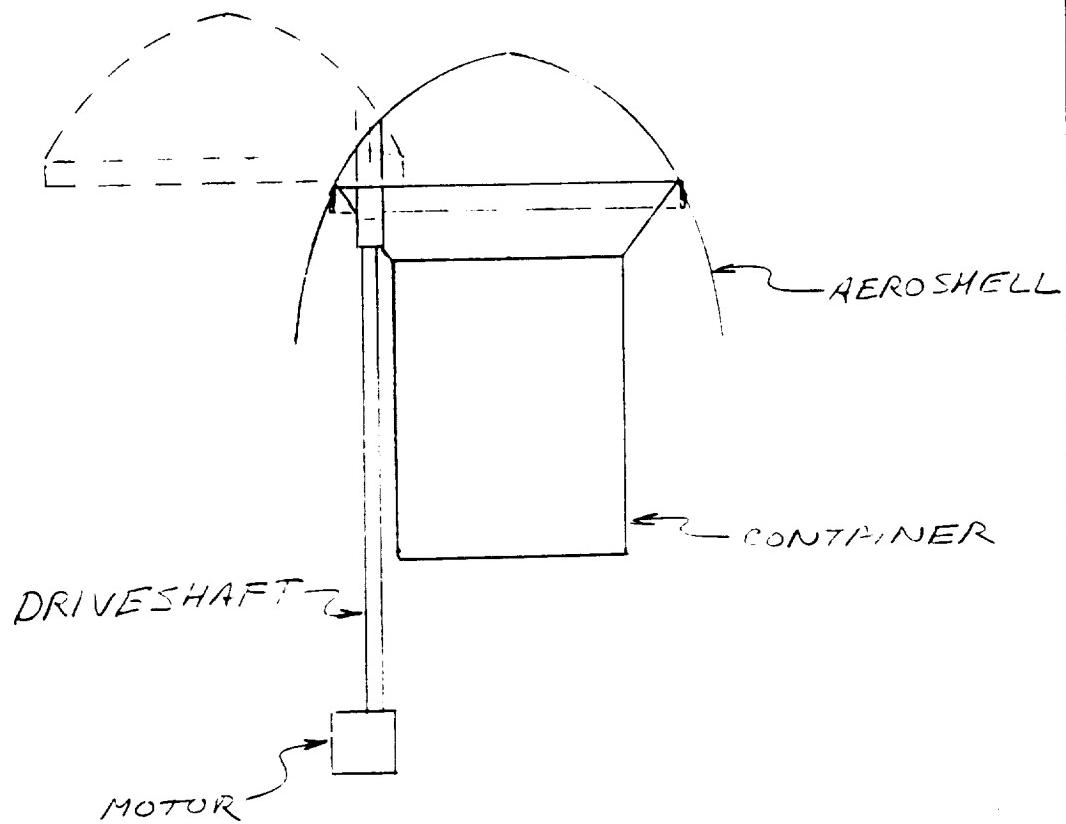


1.5m MAX.

OPTIONAL DRAFT IS  
OF PAPER QUALITY

FIG. E10

CONTAINER COVER



$$F = PR \text{ (length)} \text{ (scoop cross sect. area)}$$

where: scoop area = 4 st, and length, L = 16.5 cm, s = 10.16 cm,  
and t = 0.010 in = 0.0254 cm,

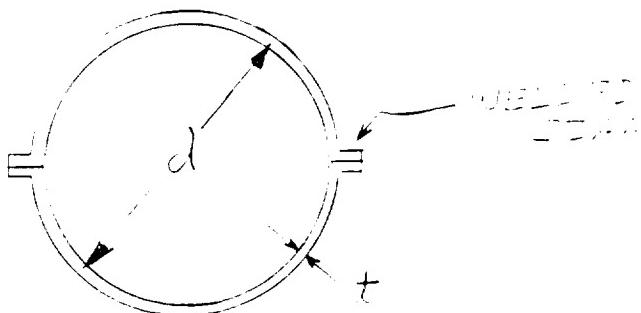
$$\text{so, } F = (PR) (L) [4 (s) (t)] = 6 (16.51) [4 (10.16) (0.0254)] \\ = 102.26 \text{ N}$$

A force of  $F = 200$  N will be used in the design of the sampler apparatus to insure an adequate safety margin.

### Mast

The dimensions of the mast (boom) will be governed by maximum compressive stress, Euler buckling, and bending. The bending requirement will be to lift the scoop, sample, and mast into the "dump position."

Maximum length of mast = 1.5m



The wall thickness of the beryllium mast is taken to be 0.0508mm, which is common for the welded seam type masts.

$$t = 0.0508 \text{ mm} = 5.08 \times 10^{-5} \text{ m} \\ = 0.002 "$$

In order to find the minimum diameter, maximum compressive stress is considered:

$$\text{stress}_y = F/A \Rightarrow F = \text{stress}_{\text{yield}} A = \text{stress}_y \pi d t$$

$$\Rightarrow d = F / (\text{stress}_y \pi t) = 200 / (3.24 \times 10^8) \pi (5.08 \times 10^{-5})$$

$$= 0.00387 \text{ m}$$

so,  $d = 0.387 \text{ cm}$

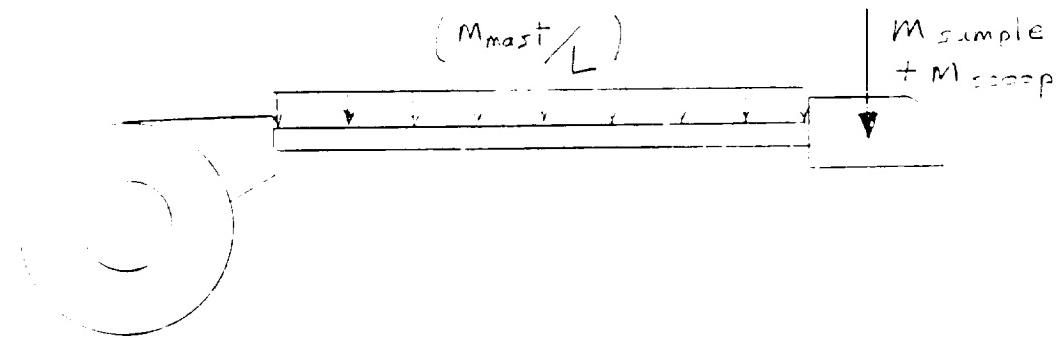
Now consider Euler Buckling:  $F = \frac{\pi^2 EI}{L^2} = \frac{\pi^2 E \pi r^3 t}{L^2}$

$$\text{so, } r = \left( \frac{FL^2}{\pi^3 Et} \right)^{1/3} = \left( \frac{200(1.5)^2}{\pi^3 (2.896 \times 10^{-6})(5.08 \times 10^{-5})} \right)^{1/3}$$

$$= 0.00995 \text{ m}$$

$$d = 0.01991 \text{ m} = 1.991 \text{ cm} = 0.784 \text{ "}$$

and finally bending, from strength of materials:  $\text{stress} = \frac{My}{I}$ ,



Mass of Be mast of length 1.5m.

$$m_{\text{mast}} = \pi d L \rho_{\text{Be}} = \pi d (1.5) (5.08 \times 10^{-5}) (1826) = 0.437 d$$

Now, the moment,  $M = m_{\text{mast}} (1/2 L) + m_{\text{sample}} L + m_{\text{scoop}} L$ .

$$m_{scoop} = 4(.1016)(.000254)(.1651)\partial_{Ti} = 4621 \text{ kg/m}^3 + m_{\text{worm gear motor}}$$

$$m_{\text{worm gear motor}} = .45 \text{ kg}$$

$$\begin{aligned} \text{so } m_{scoop} &= 4(.1016)(.000254)(.1651)(4621) + .45 \text{ kg} \\ &= .0788 \text{ kg} + .45 \text{ kg} \end{aligned}$$

$$\text{so } M = .437d(.75) + (1.5) + .5288(1.5) = .3277d + 2.2932$$

$$\text{and } I = \pi r^3 t = (\pi/8)d^3 t$$

$$\begin{aligned} \text{so } \text{stress}_y &= My/I = (.3277d + 2.2932)^4 / \pi d^2 (5.08 \times 10^{-5}) \\ &= (1.31d + 9.1728) / (1.596 \times 10^{-4}d^2) \end{aligned}$$

but  $\text{stress}_y = 3.24 \times 10^8$ , so, rearranging, we have

$$5.171 \times 10^4 d^2 - 1.31d - 9.1728 = 0$$

$$\text{or } d^2 - 2.533 \times 10^{-5}d - 1.774 \times 10^{-4} = 0$$

solving for d:

<u>d</u>	<u>F(d)</u>
.01991	.0002185
.05	.00232
.01	-.000077653
.015	.00004722

so, for bending  $0.010m < d < .015m$ . The Euler buckling in compression is the limiting case, therefore the diameter is chosen to be  $d = .01991 \text{ m}$ .

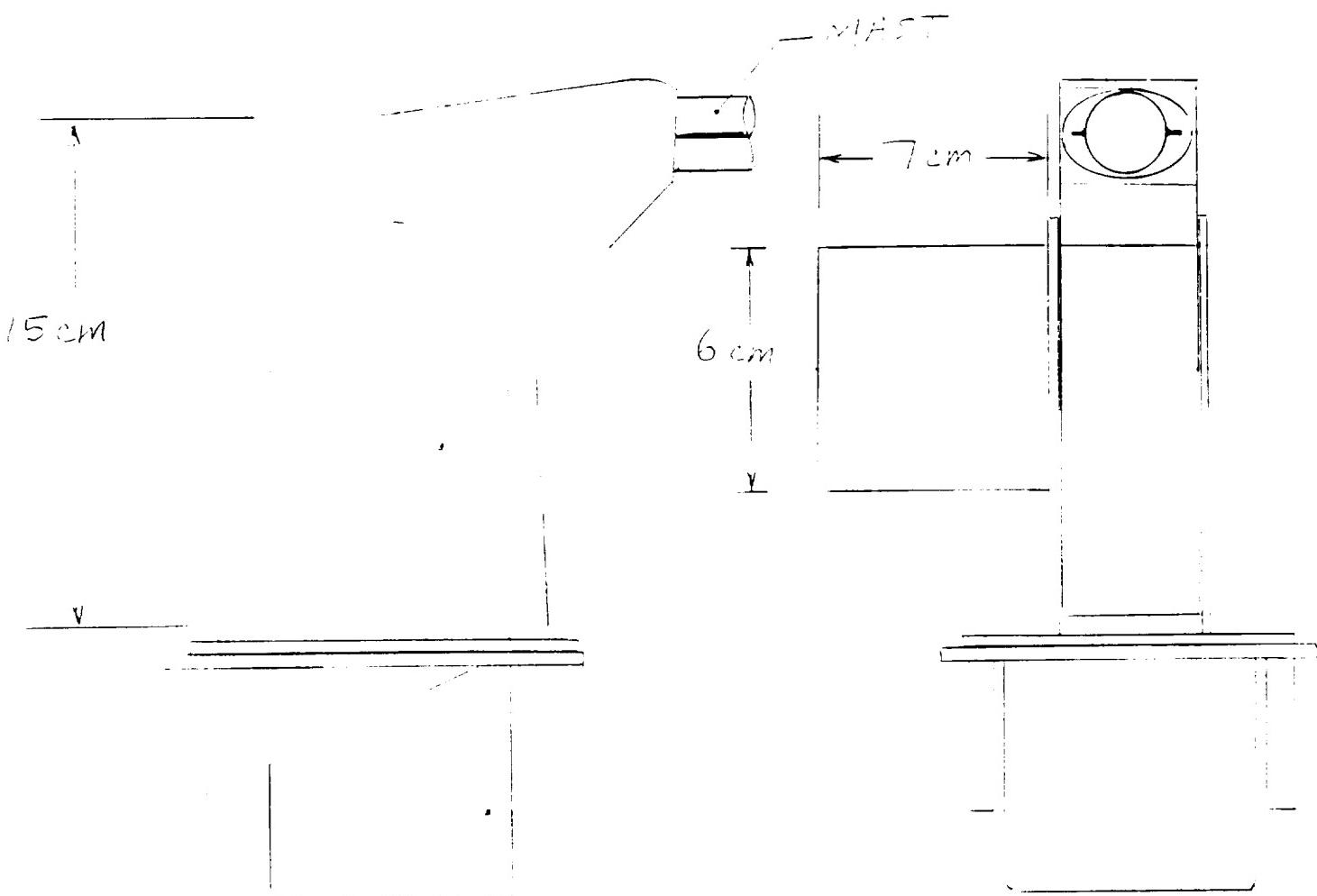
Finally, an analysis of crippling failure:

$$F_{\text{crip}}/\pi dt = 0.3E^2 t/d$$

$$F_{\text{crip}} = 0.3\pi(2.896 \times 10^{-6})2(5.08 \times 10^{-5})^2 = 1408 \text{ N}$$

The apparatus is designed for a maximum of 200N, so there is no problem here.

### Reel Assembly



THIS PAGE IS  
INTENTIONALLY LEFT BLANK

RECORDED BY  
PRADEEP KUMAR  
DATE 20-07-2012  
12 R.P.M.

Worst case torque required would occur while forcing scoop into soil, so

$$T = Fr \text{ where } F = 200 \text{ N}$$

$$r = 7.25 \text{ cm} = 0.0725 \text{ m}$$

$$\begin{aligned} T &= 200(0.0725) = 14.5 \text{ Nm} \\ &= 128.3 \text{ in lbs} \end{aligned}$$

A Hurst Instrument Motor #2602-001 has a mass of 0.68 kg and delivers 250 in lbs of torque at 1 RPM, which corresponds to a scoop speed of  $1.5 \text{ cm/sec}$ . The maximum power requirement of each motor,  $P = 54$  watts, so the electrical power requirements per sample collection are:

time for unreeling at  $1.5 \text{ cm/sec} \approx 100 \text{ sec}$ ,  $2(100) = 200 \text{ sec}$ ,  
motors running = 3,  $\Rightarrow 32400 \text{ watt-sec}$ ,  
time to rotate through boom swing  $\approx 30 \text{ sec}$ ,  $2(30) = 60 \text{ sec}$ ,  
motors running = 3,  $\Rightarrow 9720 \text{ watt-sec}$ ,  
time to rotate to dump position at 1 RPM  $\approx 15 \text{ sec}$ ,  $2(15) = 30 \text{ sec}$ ,  
motors running = 4,  $\Rightarrow 6480 \text{ watt-sec}$ ,  
time to dump  $\approx 15 \text{ sec}$ ,  
motors running = 4,  $\Rightarrow 3240 \text{ watt-sec}$ .

So, the total energy required per try  $\approx 51840 \text{ watt-sec}$ .  
The lander will have 300 watts available for 10 minutes (600 sec) = 180000 watt-sec. Therefore there will easily be enough energy for at least 3 complete attempts.

## References

### Parachute

- 1) Eckstrom, C. V. and J. S. Preisser. "Flight Test of a 30-Foot Nominal Diameter Disk-Gap-Band Parachute Deployed at a Mach Number of 1.56 and a Dynamic Pressure of 11.4 Pounds Per Square Foot." TM X-1451, Sept. 1967, NASA.
- 2) Ewing, E. G., H. W. Bixby and T. W. Knacke. "Recovery Systems Design Guide." U.S. Air Force AFFDL-TR-78-151, December, 1978.
- 3) Cockrell, D. J. "The Aerodynamics of Parachutes." NATO's Advisory Group for Aerospace Research and Development AGARDograph No. 295, July, 1987.
- 4) Peterson, C. W. "High-Performance Parachutes." Scientific American, May 1990.
- 5) Gillis, C. L. "Deployable Aerodynamic Decelerators for Space Missions." AIAA Journal of Spacecraft and Rockets, Vol 6, No 8, August 1969.

### Heat Shield

- 6) Bruhn, E. F. *Analysis and Design of Flight Vehicle Structures*. Indiana: Jacobs Publishing Inc., 1973.
- 7) Ugural, A. C., and S. K. Fenster. *Advanced Strength and Applied Elasticity*. New York: Elsevier Science Publishing Co., Inc., 1987.
- 8) Agrawal, B. N. "Design of Geosynchronous Spacecraft." Prentice-Hall, Inc., 1986.
- 9) Preliminary minimum sizing "rule of thumb" for honeycomb core density for shear strength at Hughes Aircraft.
- 10) Roark, R. S. *Formulas for Stress and Strain*, 4th ed. McGraw Hill.
- 11) ASME ME Magazine, Materials Reference Issue, 1988.

## **Soil Sample Collection**

- 12) Rennie, B. B. "New Closed Tubular Extendable Booms." Proc. Second Aerosp. Mech. Symp., Herzl, G. G., ed., JPL TM 33-355, August 15, 1967.
- 13) Herzl, G. G. *Tubular Spacecraft Booms (Extendible, Reel Stored)*, Volume 2. California: Published by author, 1970.
- 14) Meyer, R. X. "Planet Mars." Class handout, Spring Quarter 1990.
- 15) N81-10101, 318pp, Viking '75 spacecraft design. NASA Reference Publications from UCLA Engineering Library.
- 16) N84-35248, 545pp, Mars exploration, Mars data; Mariner, Voyager, Viking spacecraft. NASA Reference Publications from UCLA Engineering Library.
- 17) "Viking Exploration of Mars." National Geographic Magazine, Vol 151, No 1, Jan, 1977.

## CONCLUSIONS

We believe our conceptual design to represent the best version of the three being presented this quarter. This design has by far the lightest weight and shows much evidence of the optimization procedures we used to improve our designs while the competing designs were busy attempting to incorporate some of the best features of our earlier models. Although our design is by no means perfect and optimized in every possible way, it certainly represents a very solid foundation upon which a final design may eventually be built. Our team has enjoyed this project very much and we all feel we have learned a great deal while working on it and we look forward to working on similiar challenging projects in the world outside of UCLA.